372p

X63 81153
Code 5a

MISSILE AND SPACE VEHICLE DEPARTMENT

PROJECT APOLLO DATA BOOK

VOLUME III

[redacted]

CLASSIFICATION CHANGE

(NASA-CR-136270) PROJECT APOLLO DATA BOOK, VOLUME 3 (General Electric Co.) 372 p

N74-70391

Unclas
00/99 23899

Available to NASA Offices and NASA Centers Only.

GENERAL ELECTRIC

[redacted]

[redacted]

[redacted]

This document contains 383 numbered pages.
Copy number 131 of 150 copies.

14 MARCH 1961

PROJECT APOLLO
DATA BOOK
VOLUME III

CLASSIFICATION CHANGE
TO - UNCLASSIFIED [redacted]
By authority of E.O. No. 11652
Changed by [signature] Date 11/15/73

Prepared for:

National Aeronautics and Space Administration
Space Task Group

on contract NAS-5-302

GENERAL ELECTRIC
MISSILE AND SPACE VEHICLE DEPARTMENT
A Department Of The Defense Electronics Division
3198 Chestnut Street, Philadelphia 4, Penna.

[redacted] [redacted]

VOLUME III

TABLE OF CONTENTS

Chapter		Page
XI	LIFE SUPPORT	XI-1
	1. Introduction	XI-1
	2. Seating and Restraint	XI-2
	Design Criteria	XI-2
	Seat Type Selection	XI-6
	Restraint System	XI-8
	3. Secondary Pressure Protection System	XI-10
	Systems Analysis	XI-10
	Summary	XI-16
	4. Food and Water	XI-17
	Food	XI-17
	Water Supply	XI-30
	5. Waste	XI-33
	Solid Collection	XI-33
	Urine Collection	XI-35
	Conclusions	XI-36
	6. Survival Equipment	XI-37
	7. Lighting	XI-40
	Requirements	XI-40
	Illumination Sources	XI-41
	Crew Station System Design	XI-44
	Mission Module Area Lighting	XI-49
	Future Design Considerations	XI-50
	8. Recreation	XI-51
	9. Acoustical Environment	XI-55
	Static Firing and Launching	XI-55
	Re-entry	XI-66
	Conclusions	XI-66
	10. Atmosphere Selection	XI-68
	Introduction	XI-68
	Selection of the Atmosphere	XI-68
	11. Environmental Control System Concepts	XI-80
	Introduction	XI-80
	Effective Temperature Control System	XI-80
	Gaseous Control System	XI-83
	Ion Balance	XI-85
	Particulate Mater and Bacteria Control	XI-85
	Atmospheric Sending and Electrical Control System	XI-85
	Fire Detection and Control System	XI-86
	Leak Detection System	XI-86
	Effects of Modularization	XI-87

TABLE OF CONTENTS (Continued)

Chapter		Page
	12. Oxygen Supply	XI-90
	High Pressure Storage	XI-91
	Liquid Storage	XI-96
	Supercritical Storage	XI-96
	Hydrogen Peroxide Storage	XI-97
	Potassium Superoxide Storage	XI-97
	Trade-off Factors	XI-97
	13. Diluent Supply	XI-99
	Diluent Supply Storage	XI-99
	Nitrogen Storage	XI-99
	Summary	XI-104
	14. Carbon Dioxide Removal System	XI-106
	15. Water Vapor Control	XI-110
	Silica Gel	XI-110
	Heat Exchanger	XI-112
	16. Noxious and Toxic Gas Control	XI-114
	17. Particulate Matter Control	XI-116
	Particulate Matter Sources	XI-116
	18. Ion Control	XI-122
	Introduction	XI-122
	Ion Generator Description	XI-123
	Operation	XI-124
	Other Variable Considerations	XI-124
	19. Thermal Control	XI-127
	Cabin Thermal Control	XI-127
	20. Fire Detection and Control	XI-139
	Philosophy	XI-139
	Detection	XI-139
	Control	XI-141
	21. Sensing and Control Signal Generation for Cabin Atmosphere	XI-146
	Introduction	XI-146
	Mass Spectrometer Technique	XI-147
	Infrared and Magnetic Susceptibility Technique	XI-150
	Gas Chromatography Technique	XI-152
	Comparison of Techniques	XI-152
	Priority Telemetering of Gas Analysis During Emergency	XI-152
XII.	INSTRUMENTATION AND COMMUNICATIONS	XII-1
	1. Instrumentation	XII-1
	Vehicle Instrumentation	XII-1
	Mission Instrumentation	XII-3
	Communications	XII-14

TABLE OF CONTENTS (Continued)

Chapter		Page
XIII	APOLLO ON-BOARD PROPULSION	XIII-1
	1. Introduction	XIII-1
	2. Propulsion Phases Analysis	XIII-2
	Abort Propulsion	XIII-2
	Mid-Course Correction	XIII-5
	Lunar Orbit and Orbit Escape	XIII-7
	Attitude Control	XIII-9
	3. Mission Propulsion Analysis	XIII-10
	Lunar Orbit Mission	XIII-10
	Circumlunar Mission	XIII-18
	Lunar Landing Mission	XIII-19
	4. Reliability	XIII-20
	5. Future Efforts	XIII-24
	6. Propulsion System Considerations	XIII-25
	Propulsion System Selection	XIII-25
	Propellant Tankage	XIII-28
	D-2 Main Rocket Engine Discussion	XIII-29
	Advanced (Reverse-Flow) Rocket Engine Discussion	XIII-31
	Alternate Propellant Considerations	XIII-32
XIV	GROUND SYSTEMS	XIV-1
	1. General	XIV-1
	Factory Area	XIV-1
	Field Hangar Area	XIV-2
	Hangar Launch Pad Transporting Equipment	XIV-2
	Explosion Installation Area	XIV-2
	Field Launch Area	XIV-2
	Search and Recovery Areas	XIV-2
	Ground Tracking and Communication Areas	XIV-3
	2. Operational Sequence	XIV-4
	Factory Area	XIV-4
	Field Hangar Area	XIV-4
	Launch Pad-Flight Readiness Firing Tests	XIV-8
	Field Hangar-Final Hangar Checkout	XIV-9
	Explosives Installation Area	XIV-10
	Launch Pad-Launch Operation	XIV-10
	Search and Recovery Areas	XIV-10
	Ground Tracking and Communication Area	XIV-12

TABLE OF CONTENTS (Continued)

Chapter		Page
	3. Search and Recovery	XIV-13
	Primary Prepared Ground Landing Site (Probably The Most Desirable Recovery)	XIV-14
	Secondary Prepared Ground Landing Sites	XIV-14
	Unprepared Ground Landing Sites	XIV-15
	Calm Sea	XIV-15
	Heavy Sea	XIV-16
	Helicopters	XIV-16
	Specialized Retrieval Concepts	XIV-18
	4. Facilities	XIV-19
XV	ELECTRICAL POWER	XV-1
	1. Preliminary System Comparison	XV-1
	2. Solar Photovoltaic — Fuel Cell System Design	XV-7
	3. Appendix I	XV-12
XVI	VEHICLE DESIGN	XVI-1
	1. Introduction	XVI-1
	2. Semi-Ballistic Type Vehicles	XVI-2
	Summary	XVI-2
	Re-entry Vehicle Configurations	XVI-3
	Space Vehicle Arrangement Studies for Semi-Ballistic Re-entry Vehicles	XVI-13
	Integral Versus Modular Weight Comparison	XVI-29
	Conclusions of Semi-Ballistic Vehicle Studies	XVI-37
	Selected Semi-Ballistic Configuration	XVI-38
	Structural Description and Design Criteria	XVI-44
	Summary of Structural Characteristics	XVI-62
	Weight Analysis	XVI-64
	3. Glide Vehicle Design	XVI-71
	Summary	XVI-71
	Selected Glide Configuration	XVI-76
	Glide Vehicle Structure and Thermal Protection Materials	XVI-87
	Weight Analysis	XVI-120
	Summary	XVI-125
	4. Comparison of D-2 and R-3 Vehicles	XVI-126

LIST OF ILLUSTRATIONS

Figure		Page
XI-2-1	Mock-up of seat chosen for APOLLO vehicle	XI-7
XI-3-1	Decompression time versus hole size	XI-13
XI-4-1	Nutrient weight versus mission duration — maximum acceptability menu (3300 kilo-cal/man-day)	XI-20
XI-4-2	Nutrient weight versus mission duration — intermediate acceptability menu (3300 kilo-cal/man-day)	XI-21
XI-4-3	Nutrient weight versus mission duration — minimum acceptability menu (2500 kilo-cal/man-day)	XI-21
XI-4-4	Nutrient weight versus mission duration — for maximum, intermediate, and minimum menu acceptabilities	XI-22
XI-4-5	Nutrient volume versus mission duration — maximum acceptability menu (3300 kilo-cal/man-day)	XI-23
XI-4-6	Nutrient volume versus mission duration — intermediate acceptability menu (3300 kilo-cal/man-day)	XI-24
XI-4-7	Nutrient volume versus mission duration — minimum acceptability menu (2500 kilo-cal/man-day)	XI-25
XI-4-8	Nutrient volume versus mission duration — maximum, intermediate, and minimum menu acceptabilities	XI-26
XI-4-9	Nutrient weight versus mission duration — maximum, intermediate, and minimum menu acceptabilities	XI-28
XI-4-10	Nutrient volume versus mission duration — maximum, intermediate, and minimum menu acceptabilities	XI-29
XI-4-11	Total weight required to supply one pound of water per day versus mission duration	XI-32
XI-5-1	Pneumatic feces collection system	XI-34
XI-7-1	Brightness versus time for General Electric electro-luminescent panel, green, 160 V, RMS, sine wave cps	XI-45
XI-7-2	Brightness versus time for General Electric electro-luminescent panel, green, 120 V, RMS, sine wave cps	XI-46
XI-8-1	Vehicle weight increase versus watthours	XI-53
XI-8-2	Weight of power supplies versus watthours/day for 3-man, 14-day mission	XI-54
XI-9-1	Measured external and internal environment, Mercury space capsule	XI-58
XI-9-2	Measured external and internal environment, Big Joe capsule	XI-59
XI-9-3	Transmission loss versus frequency, single homogeneous panel	XI-61
XI-9-4	Transmission loss versus frequency, double-wall versus single-wall construction	XI-62
XI-9-5	Estimated external noise, aerodynamic boundary layer turbulence	XI-63

LIST OF ILLUSTRATIONS (Continued)

Figure		Page
XI-9-6	Effect of Mach number upon aerodynamic boundary layer noise during powered flight, APOLLO vehicle	XI-64
XI-9-7	Measured internal data, Big Joe capsule powered flight	XI-65
XI-9-8	External noise prediction for boundary layer and wake excitation, APOLLO vehicle	XI-67
XI-10-1	Atmospheric properties and blower power as a function of atmospheric composition and pressure	XI-70
XI-10-2	Flammability of paper strips versus total pressure at different oxygen partial pressures	XI-72
XI-10-3	Leakage weight penalties, nitrogen (N_2) versus helium (H_e)	XI-74
XI-10-4	Orifice area (sq ft) versus time (sec) for oxygen to reach the minimal partial pressure of 90 mm Hg considering both nitrogen and helium as the possible diluent gas at a total pressure of 7 psia and a free cabin volume of 600 cu ft	XI-75
XI-10-5	Leakage weight (lbs/hr) penalties versus hole size for helium-oxygen and nitrogen-oxygen mixtures	XI-76
XI-10-6	Weight of make-up gas supply plus blower power versus total pressure; H_e versus N_2	XI-78
XI-11-1	Hot zone requirements for crew positions in aircraft (ARDCM 80-1)	XI-81
XI-11-2	APOLLO atmospheric control system	XI-82
XI-12-1	Container weight and volume versus oxygen pressure at temperature (T) = 68 F	XI-92
XI-12-2	Compressibility factor and volume versus oxygen pressure at temperature (T) = 68 F	XI-93
XI-12-3	Primary oxygen supply system weight versus mission duration	XI-94
XI-12-4	Primary oxygen supply system volume versus mission duration	XI-95
XI-13-1	Compressibility factor and volume versus nitrogen pressure (T = 68F)	XI-100
XI-13-2	Container weight and volume versus nitrogen pressure (T = 68F)	XI-101
XI-13-3	Nitrogen storage versus time	XI-105
XI-14-1	CO_2 removal system	XI-107
XI-14-2	CO_2 adsorber	XI-107
XI-14-3	Functional schematic of CO_2 adsorber sliding valve	XI-109
XI-15-1	Humidity control systems for 3-man mission	XI-111
XI-15-2	Expected range of coil surface temperatures	XI-113
XI-17-1	Size properties of airborne particulate matter	XI-117
XI-18-1	Ionizer source design and functional diagram	XI-125

LIST OF ILLUSTRATIONS (Continued)

Figure		Page
XI-19-1	Vapor compression cycle, functional schematic diagram	XI-131
XI-19-2	Cabin air-conditioning system weight comparison	XI-135
XI-21-1	Use of mass spectrometer in a system for automatically maintaining balance of normal significant atmospheric constituents	XI-149
XI-21-2	Priority telemetering of gas analysis during emergency, block diagram	XI-155
XII-1-1	Block diagram of primary airborne communication system	XII-2
XII-1-2	Block diagram of airborne television system	XII-4
XII-1-3	Lunar surface resolution versus vehicle altitude for wide-angle camera (51.5°)	XII-8
XII-1-4	Lunar surface resolution versus altitude above lunar surface (narrow-angle camera)	XII-10
XII-1-5	Maximum error in determination of relative height of lunar surface features versus vehicle altitude above lunar surface (narrow-angle camera)	XII-12
XII-2-1	Block diagram of airborne emergency communications system	XII-15
XII-2-2	Noise figure of receiving devices	XII-17
XIII-2-1	Distance versus time, abort propulsion	XIII-3
XIII-2-2	Apogee altitude versus burning time, abort propulsion	XIII-3
XIII-2-3	Propulsion system weight for abort	XIII-6
XIII-2-4	Distance from booster versus burnout time	XIII-6
XIII-2-5	Liquid propulsion weight for mid-course correction and lunar orbit and orbit escape	XIII-8
XIII-2-6	Solid propulsion system weight for lunar orbit and orbit escape	XIII-8
XIII-3-1	Lunar orbit mission; vehicle weight minus propulsion weight	XIII-13
XIII-3-2	Ideal weight versus mission velocity, lunar orbit mission	XIII-13
XIII-3-3	Propulsion system weights	XIII-17
XIII-4-1	Typical propulsion system reliability	XIII-21
XIII-6-1	Thiokol Chemical Corporation TE-146	XIII-26
XIII-6-2	Aerojet-General Corporation "Hylas"	XIII-30
XIII-6-3	Proposed advanced propulsion system	XIII-33

LIST OF ILLUSTRATIONS (Continued)

Figure		Page
XV-1-1	Nuclear system — SNAP II version	XV-5
XV-1-2	Thermal multiplier	XV-6
XV-2-1	Electrical power and distribution system — block diagram	XV-8
XV-2-2	Solar cell concentrator system	XV-10
XVI-2-1	Inboard profile, B-2 configuration	XVI-5
XVI-2-2	Inboard profile, C-1 configuration	XVI-9
XVI-2-3	Inboard profile, D-2 configuration	XVI-11
XVI-2-4a	D-1 configuration arrangement	XVI-15
XVI-2-4b	D-1 configuration outline	XVI-16
XVI-2-5a	D-2 configuration arrangement	XVI-17
XVI-2-5b	D-2 configuration outline	XVI-18
XVI-2-6a	C-1 configuration arrangement	XVI-20
XVI-2-6b	C-1 configuration outline	XVI-21
XVI-2-7	C-2 configuration arrangement	XVI-22
XVI-2-8	B-1 configuration arrangement	XVI-24
XVI-2-9a	B-2 configuration arrangement	XVI-25
XVI-2-9b	B-2 configuration outline	XVI-26
XVI-2-10a	B-2 configuration (mock-up) arrangement	XVI-27
XVI-2-10b	Configuration B-2 (mock-up) outline	XVI-28
XVI-2-11a	Alternate B-2 configuration arrangement	XVI-30
XVI-2-11b	Alternate B-2 configuration outline	XVI-31
XVI-2-12a	General arrangement, alternate B-2 configuration	XVI-32
XVI-2-12b	General arrangement outline, alternate B-2 configuration	XVI-33
XVI-2-13	General arrangement outline, alternate B-2 configuration	XVI-34
XVI-2-14	General arrangement outline, alternate B-2 configuration	XVI-35
XVI-2-15a	Selected D-2 configuration arrangement	XVI-39
XVI-2-15b	Selected D-2 configuration outline	XVI-41
XVI-2-16	Structural arrangement	XVI-45
XVI-2-17	Structural arrangement drawing	XVI-47
XVI-2-18	Structural arrangement of recovery vehicle	XVI-49
XVI-2-19	Solar collector	XVI-52
XVI-2-20	Solar collector sequence	XVI-53
XVI-2-21	Limit collapsing pressure and running load	XVI-56
XVI-2-22	Expected impacts per second per square meter of surface area with meteoroids above a given mass versus mass of meteoroid	XVI-58
XVI-2-23	Size versus mass of meteoroids	XVI-59
XVI-2-24	Meteoroid penetration analysis (single wall)	XVI-60
XVI-2-25	Meteoroid damage — predicted number of penetrations for 15-day mission	XVI-61

LIST OF ILLUSTRATIONS (Continued)

Figure		Page
XVI-3-1	Glide configurations. Left to right: R-1 high L/D, R-2 high attitude, R-3 modified lenticular	XVI-72
XVI-3-2	Inboard profile, high L/D glider	XVI-73
XVI-3-3	Modified lenticular R-3 configuration (1/10th scale model)	XVI-77
XVI-3-4	Cockpit interior of modified lenticular configuration	XVI-77
XVI-3-5	Inboard profile, R-3 modified lenticular configuration	XVI-79
XVI-3-6	General arrangement, modified lenticular configuration	XVI-81
XVI-3-7	General arrangement outline, modified lenticular configuration	XVI-83
XVI-3-8	Three-view of modified lenticular configuration	XVI-85
XVI-3-9	Double-wall construction	XVI-88
XVI-3-10	Primary structure design criteria	XVI-92
XVI-3-11	Structural arrangement, R-1 high L/D glider	XVI-93
XVI-3-12	Structural arrangement, R-3 modified lenticular	XVI-95
XVI-3-13	Primary structure weight comparison	XVI-97
XVI-3-14	Primary structure weight comparison	XVI-98
XVI-3-15	Primary structure side studies; comparison of shapes	XVI-99
XVI-3-16	Primary structure weight comparison	XVI-100
XVI-3-17	Comparison of weight saving	XVI-102
XVI-3-18	Design A characteristics	XVI-103
XVI-3-19	Heat flux versus time	XVI-106
XVI-3-20	Total heat per unit area versus time	XVI-107
XVI-3-21	Teflon on SiC foam (all heat absorbed by ablation)	XVI-109
XVI-3-22	Teflon on SiC foam (all heat absorbed by ablation)	XVI-109
XVI-3-23	Teflon on ZrO_2 foam (Teflon to $\dot{q}$ ZrO_2 at 4000 F)	XVI-110
XVI-3-24	Teflon on ZrO_2 foam (Teflon to $\dot{q}$ ZrO_2 at 4000 F)	XVI-110
XVI-3-25	Teflon to molybdenum double wall (Teflon to $\dot{q}$ molybdenum at 2700F)	XVI-111
XVI-3-26	Teflon on molybdenum double wall (Teflon to $\dot{q}$ molybdenum at 2700 F)	XVI-111
XVI-3-27	Teflon on Haynes 25 double wall (Teflon to $\dot{q}$ Haynes at 2000 F)	XVI-112
XVI-3-28	Teflon on Haynes 25 double wall (Teflon to $\dot{q}$ Haynes at 2000 F)	XVI-112
XVI-3-29	SiC on tungsten double wall (SiC protects tungsten to 3200 F)	XVI-114
XVI-3-30	SiC on molybdenum double wall (SiC protects molybdenum to 2700 F)	XVI-114
XVI-3-31	SiC on molybdenum double wall (SiC protects molybdenum to 2700 F)	XVI-115
XVI-3-32	SiC on Haynes 25 double wall (SiC protects Haynes 25 to 2000 F)	XVI-115
XVI-3-33	SiC on Haynes 25 double wall (SiC protects Haynes 25 to 2000 F)	XVI-116
XVI-3-34	Thermal protection and thickness summary (excluding aluminum primary structure)	XVI-117
XVI-3-35	Weight summary	XVI-118
XVI-3-36	Combined weight summary — primary structure and thermal protection	XVI-119

LIST OF TABLES

Table		Page
XI-3-1	Comparison of Pressure Suits and Cocoons	XI-16
XI-6-1	Survival Equipment	XI-38
XI-7-1	Light Requirements	XI-40
XI-7-2	Comparison of Illumination Sources for Crew Station Lighting	XI-48
XI-17-1	Characteristics of Fibrous Filter Materials(2)	XI-121
XI-19-1	Cryogenic Cooling System Summary	XI-134
XI-20-1	Relative Effectiveness in Fire Tests	XI-142
XI-20-2	Toxicity Comparison	XI-143
XI-20-3	Approximate Lethal Concentration for 15-Minute Exposure, PPH by Volume	XI-144
XI-21-1	Comparison of Various Gas-Sensing Instrumentation Systems	XI-153
XI-21-2	Circuit Components Data	XI-156
XII-2-1	Margin Calculations	XII-18
XII-2-2	Equipment Description	XII-21
XIII-2-1		XIII-7
XIII-3-1	Propulsion Data — Conservative System	XIII-14
XIII-3-2	Propulsion Data — 1966 Systems	XIII-15
XIII-3-3	Weight Data for Pressurized Systems	XIII-16
XIII-3-4	Comparison of Pressurized and Pumped Systems Weights	XIII-16
XIII-3-5	Circumlunar Mission Summary Data	XIII-18
XIII-3-6	Lunar Landing — Preliminary Data	XIII-19
XIV-3-1	Helicopter Data	XIV-17
XV-1-1	General system comparison	XV-2
XV-1-2	State-of-the-art of various systems	XV-3
XV-2-1	Electrical power and distribution	XV-11
XVI-2-1	Integral Vs Modular Design	XVI-36
XVI-2-2	Summary of Design Conditions	XVI-55
XVI-2-3	Summary of Structural Characteristics	XVI-63
XVI-2-4	Weight Summary, "D" Configuration	XVI-65
XVI-2-5	Weight Breakdown, "D" Configuration	XVI-65
XVI-2-6	Preliminary Weight & Balance Status (D-2 Configuration)	XVI-66

LIST OF TABLES (Continued)

Table		Page
XVI-2-7	Detailed Weight Breakdown D-2 Configuration	XVI-67
XVI-3-1	Weights Summary of R-3 Configuration	XVI-120
XVI-3-2	Detail Weight of R-3 Configuration	XVI-121
XVI-3-3	Weight Summary, R-3 Configuration	XVI-124
XVI-3-4	Weight Breakdown, R-3 Configuration	XVI-125
XVI-4-1	Configuration Comparison	XVI-127
XVI-4-2	Comparison of Recommended Configurations	XVI-128

CHAPTER XI

LIFE SUPPORT

XI. LIFE SUPPORT

1. INTRODUCTION

Life support as considered at General Electric's Missile and Space Vehicle Department (MSVD) includes the definition, design, development and integration of that vehicle-borne equipment which protects, aids, assists and permits efficient functioning of the crew throughout the mission profile. Areas of crew support responsibility include:

Seating and Restraint
Emergency Pressure Protection
Food, Water and Waste Handling
Lighting
Noise
Crew Compartment Integration
Survival Equipment
Furnishings and Equipment
Personal Equipment
Personal Hygiene and Sanitation

During the course of this program, analysis has been conducted of these areas of responsibility in an effort to define the equipments necessary to perform the required tasks. Results of the work accomplished is summarized herein.

2. SEATING AND RESTRAINT

Prior to investigating the types of seating and restraint which might be suitable for use in the APOLLO vehicle, design criteria are established. These criteria attempt to define the features which will be necessarily included in the design of any system chosen for this vehicle. These criteria are of a general nature so as to apply to all of the vehicle configurations presently under consideration.

Upon establishment of design criteria, several methods of accomplishing the task are briefly analyzed and compared, and the most suitable selected.

DESIGN CRITERIA

The first item to be considered is the range of occupant sizes to which the system will be designed. Although the average crew size might be reasonably limited to the 75th or 80th percentile, it would be unduly restrictive to design for an individual crew member of less than the 95th percentile size. Accordingly, the seats and harnesses for the vehicle will be designed to accommodate all personnel between the 5th and 95th percentile of the flying population as defined by WADC TR 56-30.

The second point to be considered is the requirement for even distribution of acceleration loading. Total body support must be provided with emphasis placed upon elimination of local pressure points.

The system must also be such that the functions required by the crew during countdown and through all phases of flight can be readily accomplished. These include:

1. Check-out
2. Monitor functions during launch
3. Abort if required
4. Monitor vehicle health during orbit
5. Perform attitude and mid-course correction

6. Monitor and control during re-entry

7. Operate chute and landing override controls.

The existence of a multi-man crew gives rise to the desirability of providing the capability of any one of the crew members to assume the duties of any other should incapacitation occur. This can be accomplished in two ways. The first is near total duplication of instrumentation and control at any two crew stations. This, however, is obviously heavy and also demanding upon the panel and console space available. The second method is to permit crew interchange through the use of a seat/restraint system which will accommodate any one of the crew. This approach appears to be much lighter, more flexible and more attractive in other respects such as maintenance and last minute changes in flight personnel. Therefore, the requirement for a "universal" acceleration protection system is established.

Since it is expected that some equipment might be packaged behind or below the seat, it must be readily removable or foldable to provide access to this gear or to the skin for leak control.

With respect to seat adjustment, it is not expected that tilt back will be required when considering use in a vehicle which contains a mission module. The six-degree forward launch position will be perfectly comfortable in the zero g environment for performance of monitoring, communications or mid-course correction functions. In order to accommodate all crew members, however, vertical and horizontal adjustment of the seat pan will be required.

Restraint system adjustment is required so that the universal concept of station rotation still exists. In addition, the restraint system must be designed for quick release and ready removal.

The final considerations in establishing the design criteria for this system is the loading analysis which is presented below as a function of flight phase.

1. Launch

The relatively slow onset of acceleration during the launch phase appears to offer no design constraint since it will be easily tolerated. The possible exception to this might occur at stage burnout when the energy stored in any resilient material of which the seat is made will tend to cause overshoot as it recovers. The restraint system must be so designed to prevent this overshoot and control rebound.

2. Mid-course Corrections

Very low acceleration levels of between 0.5 and 1.0 "g" are expected during this phase. No particular problem is anticipated.

3. Re-entry

Here accelerations in the eyeballs-down direction as well as eyeballs-in direction are anticipated. While the vehicle system will be designed such that this loading does not exceed human tolerance, the seat must be capable of accepting the resultant force vector throughout the angle of attack expected. The relatively slow onset and decay of acceleration in this flight phase again does not appear to offer any unusual design constraints. The restraint system, however, must be capable of pre-positioning and pre-tensioning prior to the onset of this acceleration. Since body position in the seat is important and since the zero "g" environment prior to re-entry will present some difficulty to the astronaut in attaining this position, it is felt to be mandatory that the restraint system accomplish this task for him. Pre-loading of the harness will accomplish this positioning task and will also "snug" the crewman into his seat to prevent oscillation during the early portion of the re-entry phase.

4. Parachute Opening

Here the loading magnitude is not expected to be as severe as that experienced at impact. However, the possibility of this load being applied through a moment arm caused by disorientation of the capsule attitude with respect to the vector of chute load application is real, particularly during a launch abort when the recovery vehicle might tumble. The seat/restraint system must therefore be designed to accept these

loads through a rapidly changing vector applied at any direction with respect to vehicle centerline.

5. Impact

Easily the most severe condition, impact poses the greatest problem in seat/restraint design. Nominally, the seat must accept the impact load transmitted to it through the capsule structure. This load, applied in the eyeballs-in direction, is at very high rate of onset and of short duration. Care must be exercised in design to assure that overshoot does not occur at a rate which causes "g" to exceed human tolerance. Head restraint will be required here to prevent forward motion of the head at rebound.

This nominal impact condition, however, is academic only, since any relative ground velocity caused by surface wind, oscillation under the chute or oblique terrain will result in an impact direction other than the ideal vertical and a magnitude of impact greater than the ideal descent rate.

Although the expected range of impact accelerations (10 to 20 G) are easily tolerated if landing is normal to the occupant's spine, they may become critical if imposed at shallow angles to the long axis of the spine. This therefore requires a head-restraint system which permits the head to move with the torso when subjected to an eyeballs up or down acceleration.

Recovery vehicle tumbling upon land impact and "tuck under" upon water impact are also conditions to which the seat and restraint system must be designed.

6. Abort

It is expected that higher than launch and re-entry but lower than impact accelerations will be experienced during abort. The only unusual design condition expected here is the "g" reversal which is caused by aerodynamic drag at rocket burn out. The restraint system must under these conditions accept this load and still hold the astronaut secure in the seat so that parachute opening and impact accelerations can be tolerated.

SEAT TYPE SELECTION

Based upon the constraints imposed in the above design criteria, a selection was made of the type of seat proposed for use in the APOLLO vehicle. First, it is pointed out that the requirement for "universal" occupancy and ready removability immediately rules out any further consideration of either a "Mercury"-type molded couch or a soft foam, contoured unit, since these seats are by nature individually fit.

The Micro Balloon seat was also investigated. This unit is constructed of a "mattress"-type bag filled with a powder which sets into any shape desired upon exposure to vacuum. This contour can be easily changed by repressurizing, reforming, and then re-evacuating. While interesting, this concept of seat design has several disadvantages which include its bulky nature and expected high installed weight. In addition, too little work has been done with this seat to indicate its feasibility at this time. However, its possibilities appear attractive enough to warrant additional consideration and investigation.

The Raschel Net seat developed by WADD has been investigated and appears to offer solution to all the design criteria developed above. It is lightweight, "universal", readily removable, and has in test indicated an excellent capability to distribute acceleration loads up to 20 g continuous acceleration.

In addition, it has unusually fine vibration damping characteristics. The only problem area which appears to offer significant challenge is in the resilient property of the net which tends to cause overshoot during impact or shock loading conditions. This problem, however, does not appear to require an unusually novel solution, since it is felt that the combination of an adequate restraint system and possibly a "bottoming" device will reduce this rebound to a negligible amount.

This seat, therefore, has been chosen as the one most suitable for the APOLLO mission. Figure XI-2-1 shows a Missile and Space Vehicle Dept. mock-up of this type seat. Installed weight including adjustment devices is estimated at 23 pounds.

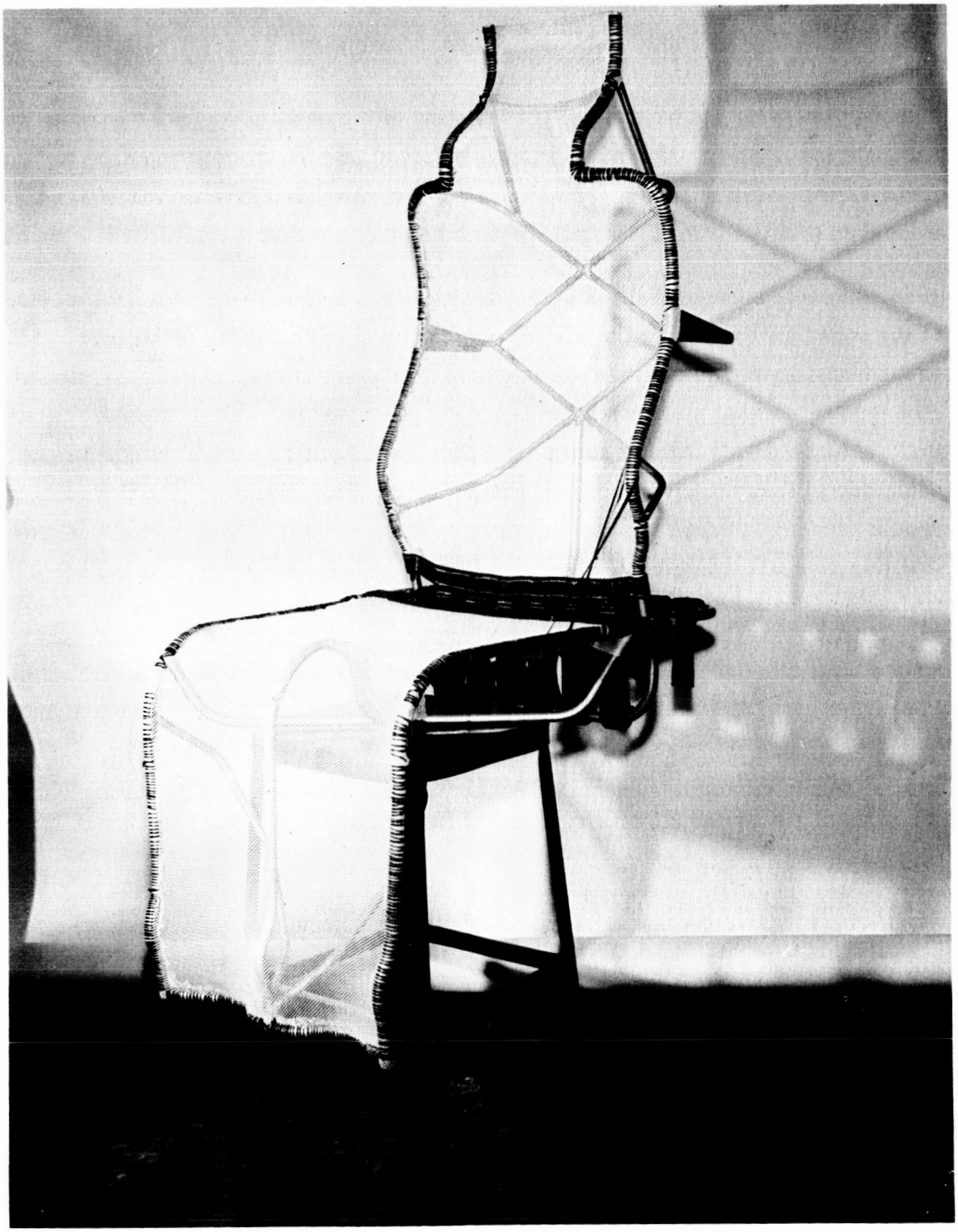


Figure XI-2-1. Mock-up of seat chosen for APOLLO vehicle

RESTRAINT SYSTEM

Although present restraint systems consisting of lap belt and shoulder harness are adequate for use by military pilots, it is believed that the system required for the APOLLO vehicle demands more positive support through a greater variety of situations than present type of aircraft-restraint systems can provide.

Unfortunately, shoulder straps cannot be depended upon to prevent lateral displacement of the upper torso. Hence, a vest-type restraint system will be mandatory. This vest will be capable not only of distributing the loads as uniformly as possible, but will also prevent lateral movement from exceeding reasonable limits. It is anticipated that the loads will be taken out at four points, i.e., one on each shoulder and one on each side of the torso. In addition, the inclusion of an inverted "V" belt at the crotch should prevent submarining and undue pressure at the abdominal area. It is expected that this vest will be fashioned specifically to fit each of the crew members being trained for the mission. Common take-out points will, however, mate to corresponding buckles at the seat so that the system retains its universal character.

Positive restraint of the extremities will also be provided. This will be accomplished by either clamps or webs placed over the knee and ankle and integrated control-wrist locks provided for the arms. The intent here is to prevent any large-scale skeletal displacement from occurring either in a direction parallel with or at acute angles to the long axis of the bones.

The prevention of relative motion between the head and upper torso is also extremely important since irreversible injury to the spinal cord can result if undue displacement occurs. Therefore, since fore and aft as well as lateral restraint of the head is necessary, care must be exercised in design to assure that up and down motion of the head with respect to the seat can exist as the torso moves in these directions under positive or negative g. Additional study is required before a satisfactory method of accomplishing this head restraint is proposed.

In addition, the system proposed offers the advantage of being easily adjustable and quickly removed should an emergency so dictate.

The conclusion which is reached is that an integrated seat/restraint system is mandatory for proper acceleration protection. The net seat and vest system proposed appears to offer all the basic ingredients for successful solution to this problem. As an example, the use of spring-driven take-up reels will provide not only pre-positioning and pre-loading but also will aid in reducing the rebound which is expected under some impact loading conditions.

The weight of the complete seat and restraint system is estimated at 30 to 35 pounds.

3. SECONDARY PRESSURE PROTECTION SYSTEM

SYSTEMS ANALYSIS

A secondary pressure protection system must provide an emergency atmosphere when either cabin pressure integrity is lost or the cabin environment becomes irrepairably contaminated. Cabin pressure integrity can be lost through structural failure caused by any one or a combination of loading, vibration, shock or meteoroid impact. A marked increase in design leak rate from any cause is also considered pressure vessel failure. The cabin atmosphere can, of course, be contaminated by the products of fire, on-fire control, radiation, materials out-gassing and arcing electrical equipment.

For the sake of simplicity it would be well to discuss this latter area first in order to eliminate it from further detail consideration. This elimination is possible merely by pointing out that the build-up of toxic gases within the cabin will be at a relatively slow rate and hence will have little effect on the design or selection of a secondary-pressurization system. That is to say that, regardless of the method of achieving a new atmosphere, the "time of action" requirement for protection against contamination can satisfactorily be accomplished. It is pointed out, as an extreme case, that the cabin could be purged of these gases at no total pressure reduction. Although this is an inefficient method from the standpoint of stored gas used, it is mentioned only to illustrate that cabin contamination protection will not be a governing criterion for the secondary pressure protection system.

One of the most important criteria however, in the design of this system, is the time available to take corrective action during a decompression. This time is defined herein as the interval during which the oxygen partial pressure drops from the nominal 180 mm Hg to the minimum required to prevent the onset of hypoxia.

From above it is seen that this decompression can occur either through structural equipment failure or meteoroid puncture. Considering these in order, it is pointed out that it appears most reasonable to assume that structural failure, if it occurs at all, will occur during the launch or re-entry phases of flight, since these are the periods

of maximum loading, vibration and shock. It is therefore assumed that the pressurization system must be such that it can be at "ready" during both of these phases. In the case of the suit, as an example, the crew would be wearing the garment hooked up with face plate closed, during these periods.

Having satisfactorily survived launch, the cabin must be assumed to be structurally sound throughout orbital flight. While it is conceivable that leaks may occur or increase in size during this portion of the mission, the elimination of virtually all but pressure loading from the cabin structure permits an extremely high level of confidence to exist with regard to pressure integrity.

We are therefore left as a design consideration that decompression which might be caused by meteoroid impact. For the purposes of this investigation the reliability goal of the secondary pressure protection system is set at 0.999. With regard to meteoroid puncture then, the system must be able to provide protection to the crew during a decompression which would be expected to occur once every 1000 missions.

With these ground rules established, it is now possible to calculate the decompression time to which the system must be designed. Using the surface area of 20 square meters for the D-2 configuration, a mission duration of 14 days, and a reliability of 0.999; 2.4×10^{10} square meter seconds are presented to the meteoroid population. The meteoroid population curve of Chapter XVI determines that the largest meteoroid expected weighs in the order of 0.23 grams.

Entering the meteoroid size curve of Chapter XVI with this number, we see that a stony (more damaging) meteoroid of this weight can be expected to be $0.085 \times 2 = 0.17$ in. diameter.

Empirical results based on tests conducted at MSVD on meteoroid bumpers indicate that the honeycomb outer skin structural arrangement will limit penetration holes to 6 times the diameter of the meteoroid since the double wall and core of the honeycomb is expected to absorb more energy than sheet material of equivalent strength.

We can therefore state that our design criterion must be that decompression which is caused by a hole of 6 x 0.17 = 1.02 in. in diameter.

Figure XI-3-1, decompression time versus hole size for the 302 cubic feet D-2 configuration and the 400 cubic feet R-3 configuration, now can be used to define "time of action". We see from this curve that a hole 1.02 inches in diameter in the D-2 pressure vessel causes reduction of oxygen partial pressure to 90 mm Hg in 0.85 minutes or 51 seconds.

Repeating these calculations for the R-3 configurations indicates that an action time of 54 seconds can be expected.

It is now necessary to discuss another design criterion which imposes constraints upon the secondary-pressure protection system. This criterion is that one which dictates the requirement for "shirt-sleeve" flying capability. As applied to the problem here, it simply indicates that the system must be designed so that the crew need not wear a pressure suit nor any other protective garment continuously. This in turn, when considering the "time of action" arrived at above, presents a limit on the use of the pressure suit since it is inconceivable that this device could be donned during the time available.

Use of the pressure suit, however, cannot be completely ruled out since the vehicle is designed with two compartments and since the "shirt-sleeve" philosophy can be

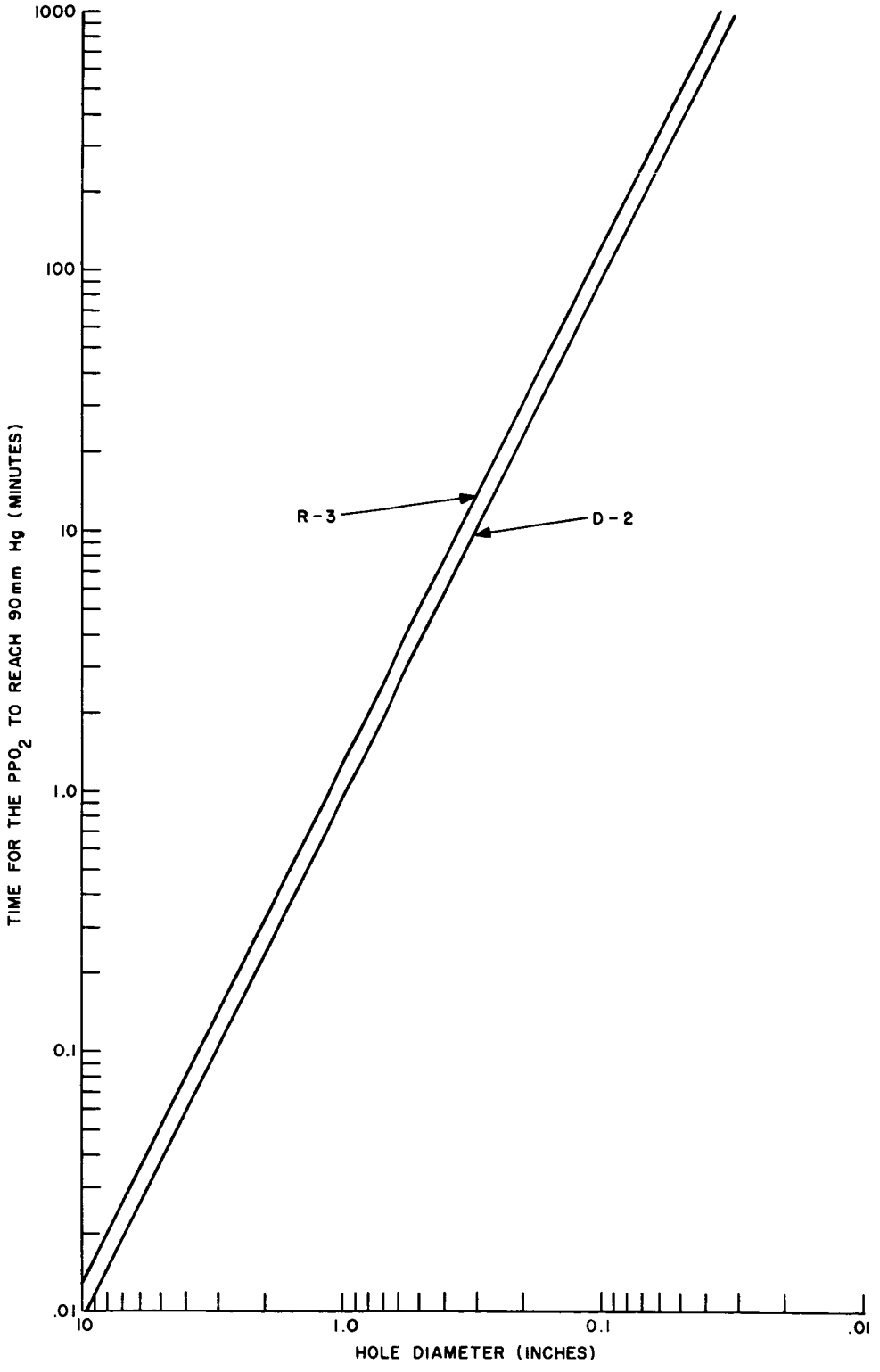


Figure XI-3-1. Decompression time versus hole size

modified to permit one member of the crew to be clothed at all times who is at rest in the R/V. Under these conditions the other two members would, at penetration of the R/V, retreat into the mission module and seal the hatch behind them. The suited crew member then attempts to repair the leak, and can, upon success, repressurize the compartment. Should the puncture be irrepairable, then the other two crewmen can don their suits under no particular time constraint, enter the R/V and make preparations for return. In short, it is proposed that even though our calculated "time of action" is on the order of one minute, the use of two compartments permits the pressure suit to satisfy the criteria established thus far. It is pointed out, however, that an airlock must be provided here in order to reduce the weight of atmosphere lost during transfer from one compartment to the other.

It is now appropriate to discuss the second method of emergency pressure protection to be analyzed. This method employs an encapsulated seat principle similar in configuration to the escape capsules used in the B-58 aircraft and scheduled for the B-70. These "cocoons", a crude example of which is displayed in the mock-up, will perform the same basic function as the suit, but will by nature, be closed rapidly so that the time to get into a seat is virtually all the time that is necessary to become protected.

This means that, upon decompression, the crew, two members of which will normally be in seats, will simply close the hood and remain inside during the course of the emergency. Now, since the requirement to attempt repair exists, one suit must be provided within one cocoon. This crewman now dons this suit and exits into the decompressed vehicle to perform the repair if possible. If repair cannot be made, then he retreats back into the "cocoon" and return is initiated.

It is pointed out that since this suit has limited use, it need not be as sophisticated as those normally considered for use in vehicles of this nature. It may, in fact, be optimized for the job to be performed, and ease of donning with leakage and long-term habitation requirements minimized. It is also pointed out that the primary vehicle controls must be arranged such that they are located within this enclosure for use during this emergency.

At this point, having considered only the "time of action" constraint, we might say that both the suit and "cocoon" approaches are suitable for this vehicle. However, other

criteria must be considered prior to making a selection. One of primary importance is that of habitability, for except for emergencies occurring at the extreme beginning or end of the mission, the time to return can be as long as 3 to 4 days. Unfortunately, little testing of long-term pressure-suit habitability has been accomplished to date. However, the data which does exist is rather discouraging for confinement times of 3 to 4 days. Briefly, the conclusions one might reach after examination of this information is that survival is probable, but performance can be expected to be severely reduced.

In addition, body waste handling and feeding for this relatively long period present certain problems. While the urine and feces containment problem could conceivably be ignored, closed face-mask feeding of the bite-sized food form which is chosen for this mission presents a problem not easily solved.

The use of the cocoon however, suggests that 1) habitability problems and performance decrement will be substantially reduced and 2) feeding and waste handling will present no unsolvable design constraint.

The cocoon has, in addition, other advantages with respect to the pressure suit. One of these is in regard to restraint. Obviously, a highly sophisticated restraint system will be required for either of the configurations under study. It can be reasonably stated that total body restraint, particularly with regard to the head, can more effectively be accomplished if the system can operate without the design restrictions imposed by pressure suit hardware.

Another area yet to be discussed is mission completion possibilities when design leakage is substantially exceeded as a result of the rigors of launch. Should this situation occur, then the low habitability of the pressure suit would render abort mandatory. The use of the cocoon however might permit mission completion by retreating into these units, sealing them and remaining there until the vehicle has reached the proximity of the Moon. Having in this manner conserved the on-board oxygen and nitrogen supplies, the cabin could be repressurized and the crew could emerge from the cocoons to perform the tasks necessary during lunar orbit. Re-entering the cocoons, the crew now can return by once again reducing the effective leakage loss.

Mission flexibility is a very basic and important design criterion for equipment under consideration for this program. The modular construction of the vehicle strongly implies the use of the R/V for any mission without necessarily combining it with a mission module. Since under this flight condition the "shirt-sleeve philosophy" will still exist and and since the "time of action" will, as a result of using only one compartment, be reduced, the pressure suit must be considered unsatisfactory. Therefore, in order to preserve the mission flexibility requirement which has been a prime consideration in vehicle design, it will be necessary to incorporate in the R/V an emergency pressurization system other than a suit. The encapsulated seat described above is, therefore, proposed.

SUMMARY

A matrix is prepared in order to aid the comparison of pressure suit and cocoons for use in the APOLLO vehicle. This matrix is presented in Table XI-3-1 below.

TABLE XI-3-1. COMPARISON OF PRESSURE SUITS AND COCOONS

Design Consideration	3 Pressure Suits + 1 Airlock	3 Cocoons + one Pressure Suit
Shirt-Sleeve Flying Capability	Fair	Good
Loss of Pressure Protection (total vehicle)	Good	Good
Habitability	Poor	Fair
Inflight Repair Capability	Good	Good
Mission Flexibility	Poor	Good
Restraint Integration	Fair	Good
Leak Rate	Good	Good
Weight	235	130

Rapid examination of this table indicates that the cocoon is in all ways as good as or superior to pressure suits for use in this vehicle. This approach has therefore been chosen as the secondary pressure protection system to be designed for the APOLLO system.

4. FOOD AND WATER

FOOD

When choosing a nutritionally adequate diet for any mission, the factors of weight, volume, and desirability are the most critical parameters to be considered. A meal that is of low weight and volume but highly palatable is required. A highly acceptable diet is considered to be extremely important in offsetting the stresses occurring during the mission.

In order to minimize weight and volume, dehydrated foods are utilized wherever possible. To increase the acceptability of the diet, canned and frozen foods also may be included. A nutritional level of over 2500 kilo calories per day has been suggested, 15 percent of which is obtained from protein, 32 percent from fat, and 53 percent from carbohydrate.

Menus have been designed for three levels of acceptability. They are maximum, intermediate, and minimum.

The maximum-acceptability diet is naturally the most desirable, with a high nutritional level and a wide variety of foods including snacks and many between-meal beverages. Appetizers are provided in canned and dehydrated forms while the main dishes and vegetables are frozen or canned. Desserts, bread, cake and various items, such as candy, nuts and sandwiches, are provided in canned or frozen forms. Beverages and soups are in the dehydrated form and are reconstituted by the addition of hot water. All the foods are precooked and can be heated before consumption. Thus, an oven is required. The canned and dehydrated foods are stored on shelves and a refrigerator is required for the frozen foods. Beverages, juices, and fruit can be made more acceptable by the provision of a chill compartment in the refrigerator. Because of the need for hot water, an oven, and a refrigerator, a fair amount of power will be consumed in implementing this diet.

The food is provided at the rate of 3.20 pounds per man day and water at 6.0 pounds per man day, 1.65 pounds of which is contained in the food.

Menus for space missions have been developed (Refer to ARDC technical report No. 60-8, July, 1960) and listed below is a representative one-day menu of a maximum acceptability diet.

Breakfast	Lunch	Supper
Orange juice	Tomato juice	Vegetable soup
Oatmeal	Roast beef with mushroom gravy	Tuna and rice
Scrambled eggs	Buttered peas	Buttered string beans
Bacon	Baked potatoes	Baked apple
Bread-butter	Bread-butter	Bread-butter
Milk	Butterscotch pudding	Chocolate-chip cookies
Coffee	Coffee or tea	Cocoa or milk

The intermediate acceptability diet differs from the maximum in that there are no frozen foods provided and some of the supplementary beverages have been deleted. This narrows the variety of foods somewhat and reduces the water consumption. However, a high nutritional level is still available and sufficient water for the metabolic processes is provided. Appetizers are provided in canned and dehydrated forms, as they are in the maximum acceptable diet, and the beverages and soups are still dehydrated. The main dishes, vegetables, and desserts are now in the canned and dehydrated forms and are not as delectable as the frozen form. The foods are precooked and should be heated before consumption. Thus, an oven is required. The dehydrated foods require the addition of hot water. Since frozen foods have been eliminated, there is no need for a refrigerator, but a means of reducing the temperature of beverages such as milk and juice can be provided by utilizing a cold region of the air conditioning system.

The food is provided at the rate of 3.08 pounds per man day and water at 5.0 pounds per man day, 1.64 pounds of which is contained in the food.

Thus, it can be seen that the weight of the food for the intermediate diet is slightly less than that of the maximum diet, and the water consumed is reduced by 1.0 pounds per man day, and the weight of the refrigerator and associated equipment has been eliminated.

The minimum acceptability diet is one that involves the use of practically all precooked dehydrated foods supplemented with sandwiches, food bars, and bite-sized solid foods. The addition of hot water is the only requirement necessary for the reconstitution of these foods. Instead of the wide variety of foods shown in the previous diets, a relatively narrow range of foods is available. The problem of food monotony, a lower level of food acceptability, and probable resulting decrease in food consumption will add to the stresses of space travel.

For such a diet, the only components which are necessary in addition to the food and water are storage compartments and a water heater. The weight, volume, and power requirements of this diet thus are reduced to a minimum. Food is provided at the rate of 2.15 pounds per day and water at 4.85 pounds, with 1.0 pound contained in the food.

The minimum acceptability diet represents a decrease of both food and water from the intermediate-acceptability diet, but does satisfy minimum nutritional requirements.

Ten graphs have been constructed to be used in estimating the weight and volume of nutrients needed for missions of up to 70 man-day duration. Figure XI-4-1 illustrates the individual weight constituents, such as food, water, refrigerator shelving, and food heater, required for missions up to 70 man-day duration for a maximum acceptability diet. Sub-totals, as well as the over-all total, are also illustrated on this graph.

Figures XI-4-2 and XI-4-3 illustrate the constituents and total weights for the intermediate and minimum acceptability diets respectively. The grand total weights of the maximum, intermediate, and minimum acceptabilities are assembled on Figure XI-4-4.

Figures XI-4-5 through XI-4-8, inc., are the volume equivalents of Figures XI-4-1, XI-4-2, XI-4-3 and XI-4-4.

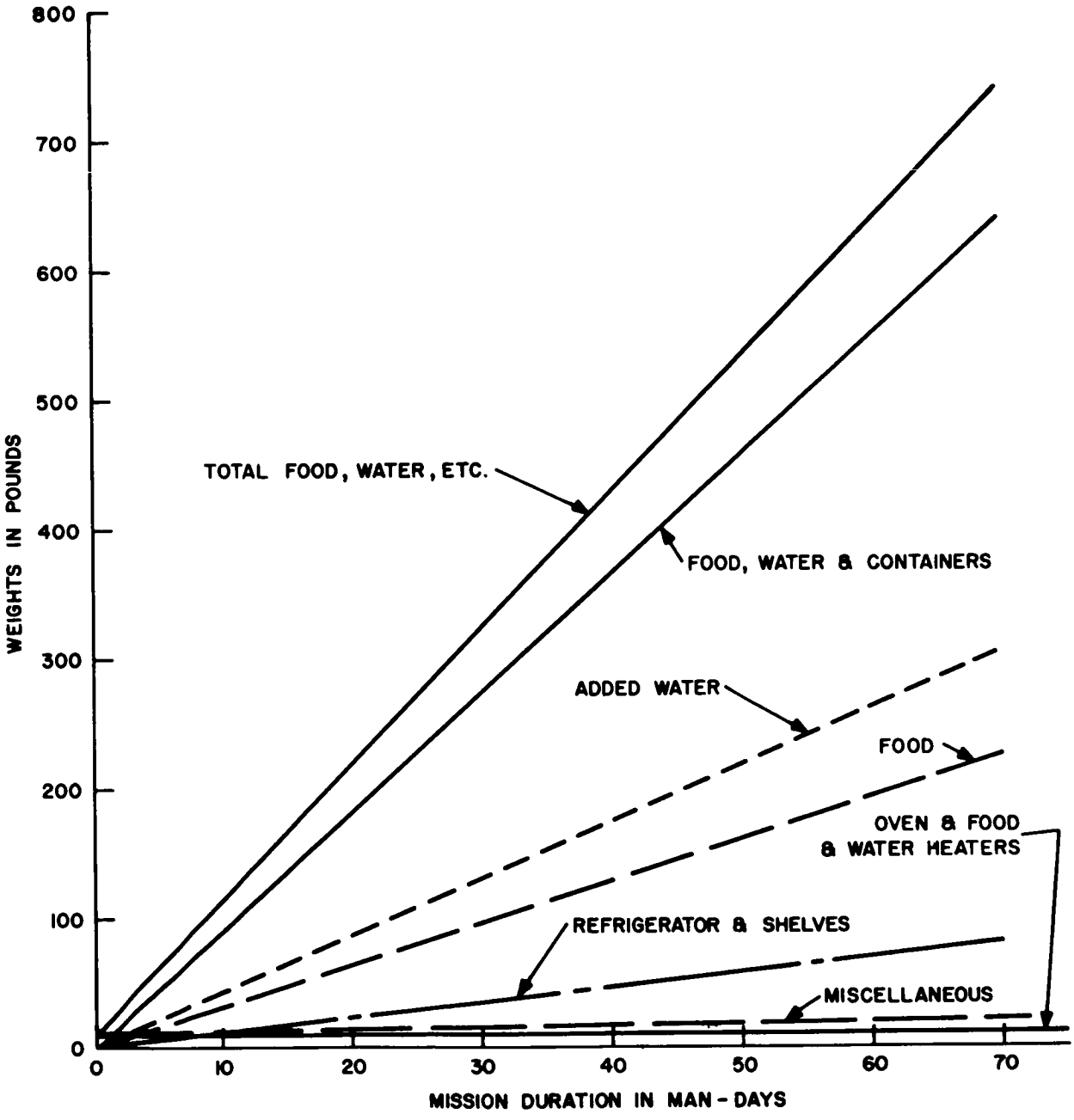


Figure XI-4-1. Nutrient weight versus mission duration — maximum acceptability menu (3300 kilo-cal/man-day)

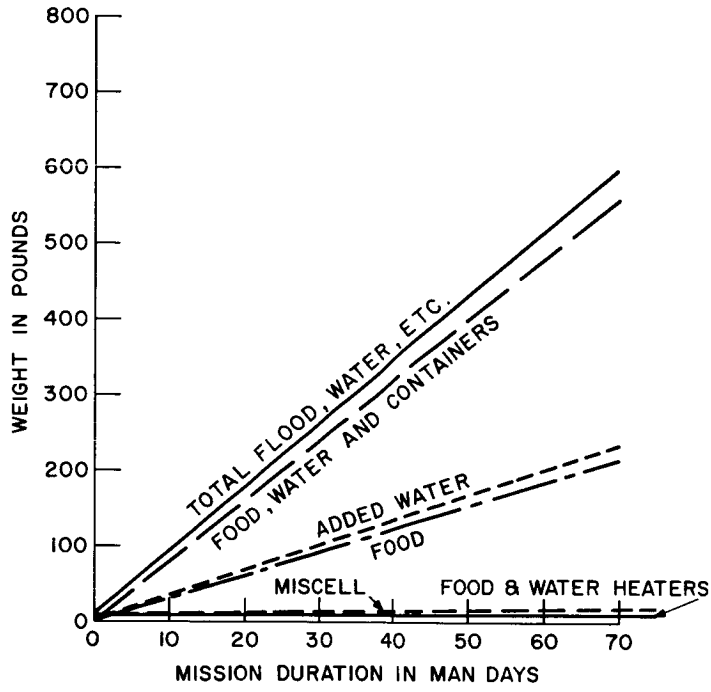


Figure XI-4-2. Nutrient weight versus mission duration — intermediate acceptability menu (3300 kilo-cal/man-day)

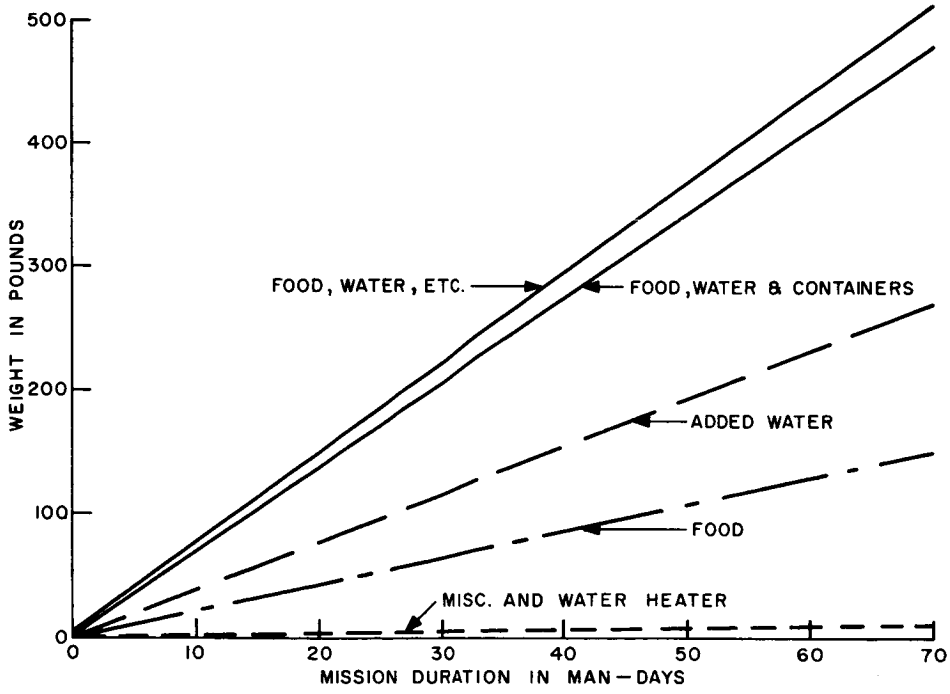


Figure XI-4-3. Nutrient weight versus mission duration — minimum acceptability menu (2500 kilo-cal/man-day)

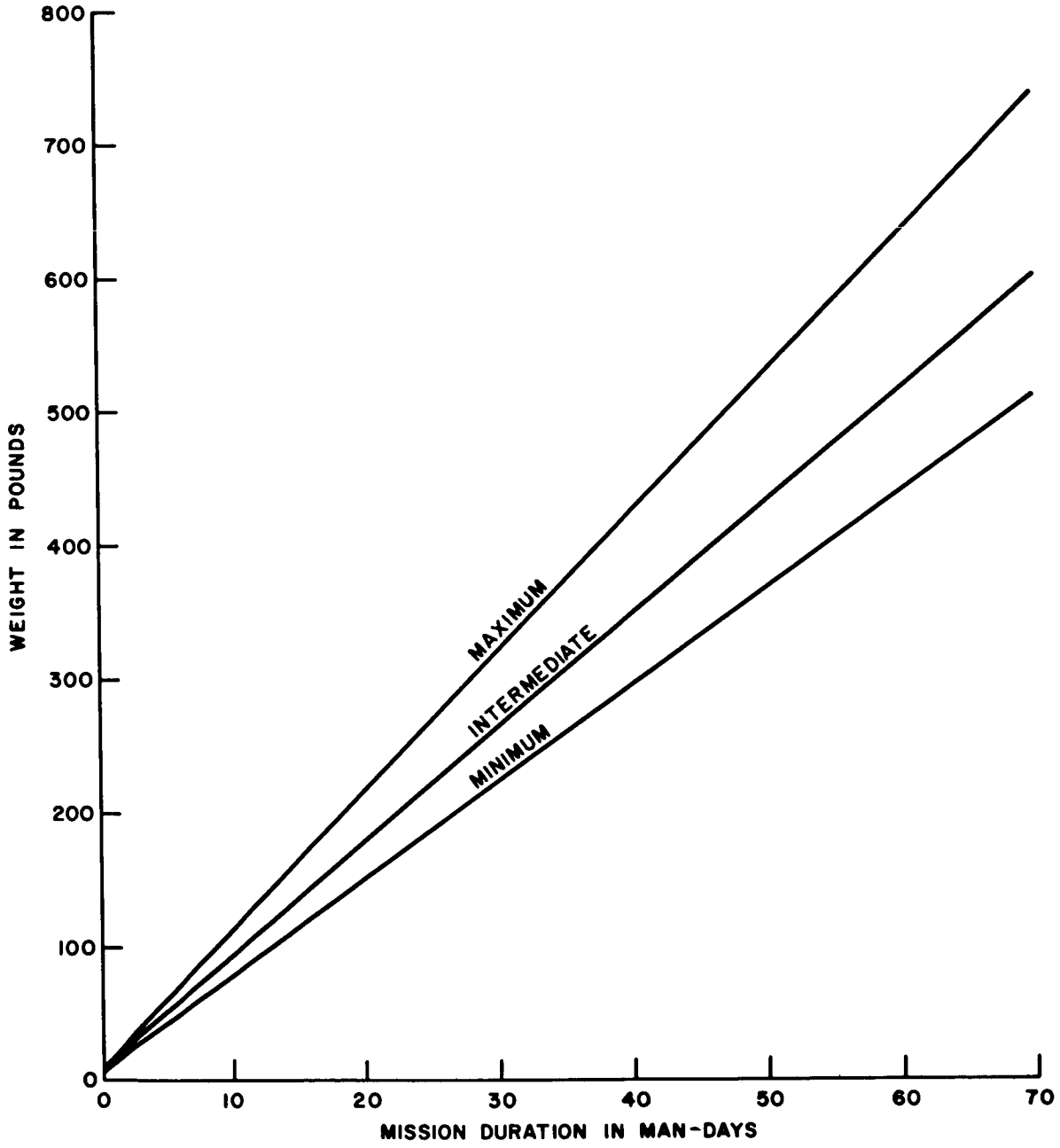


Figure XI-4-4. Nutrient weight versus mission duration — for maximum, intermediate, and minimum menu acceptabilities

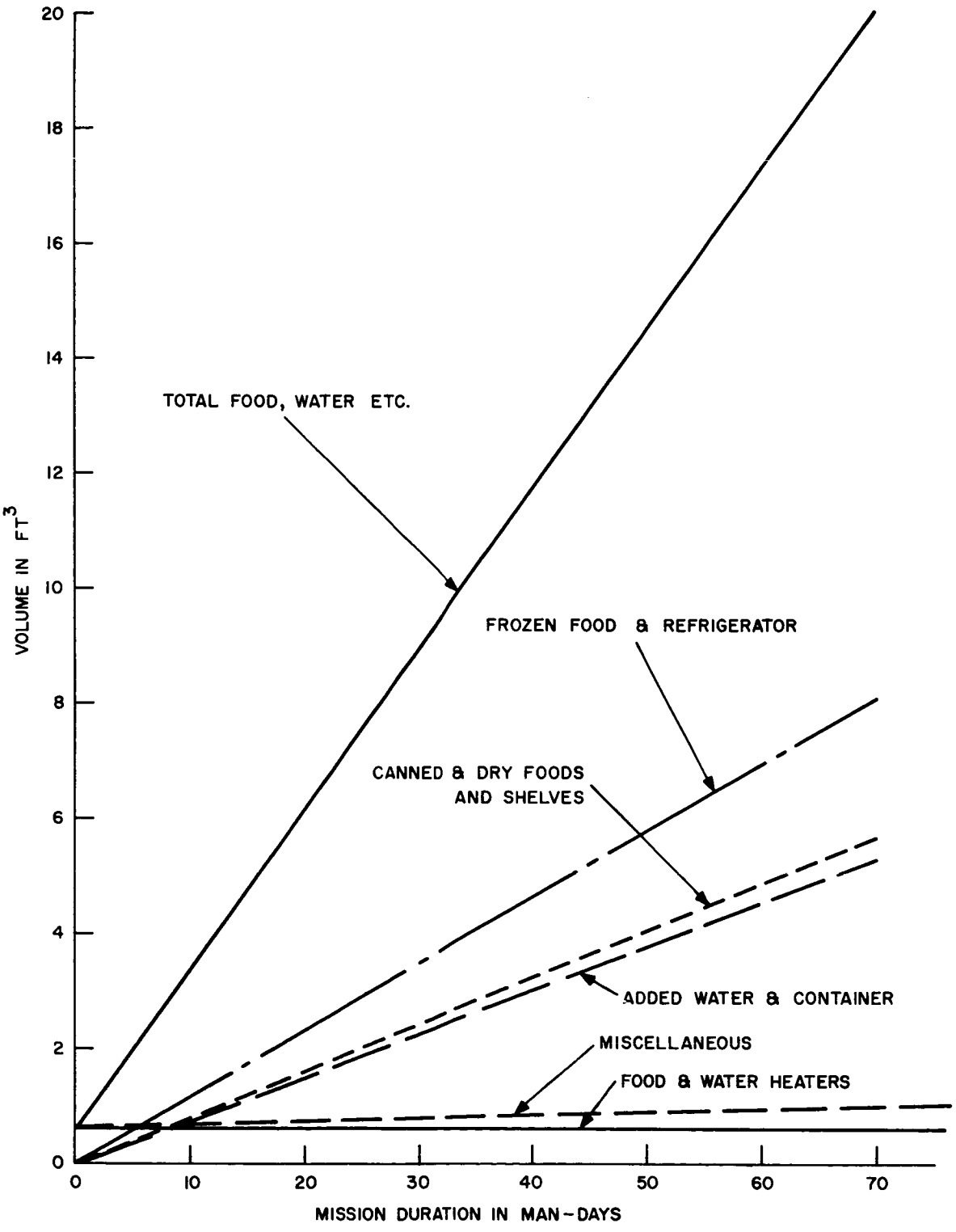


Figure XI-4-5. Nutrient volume versus mission duration — maximum acceptability menu (3300 kilo-cal/man-day)

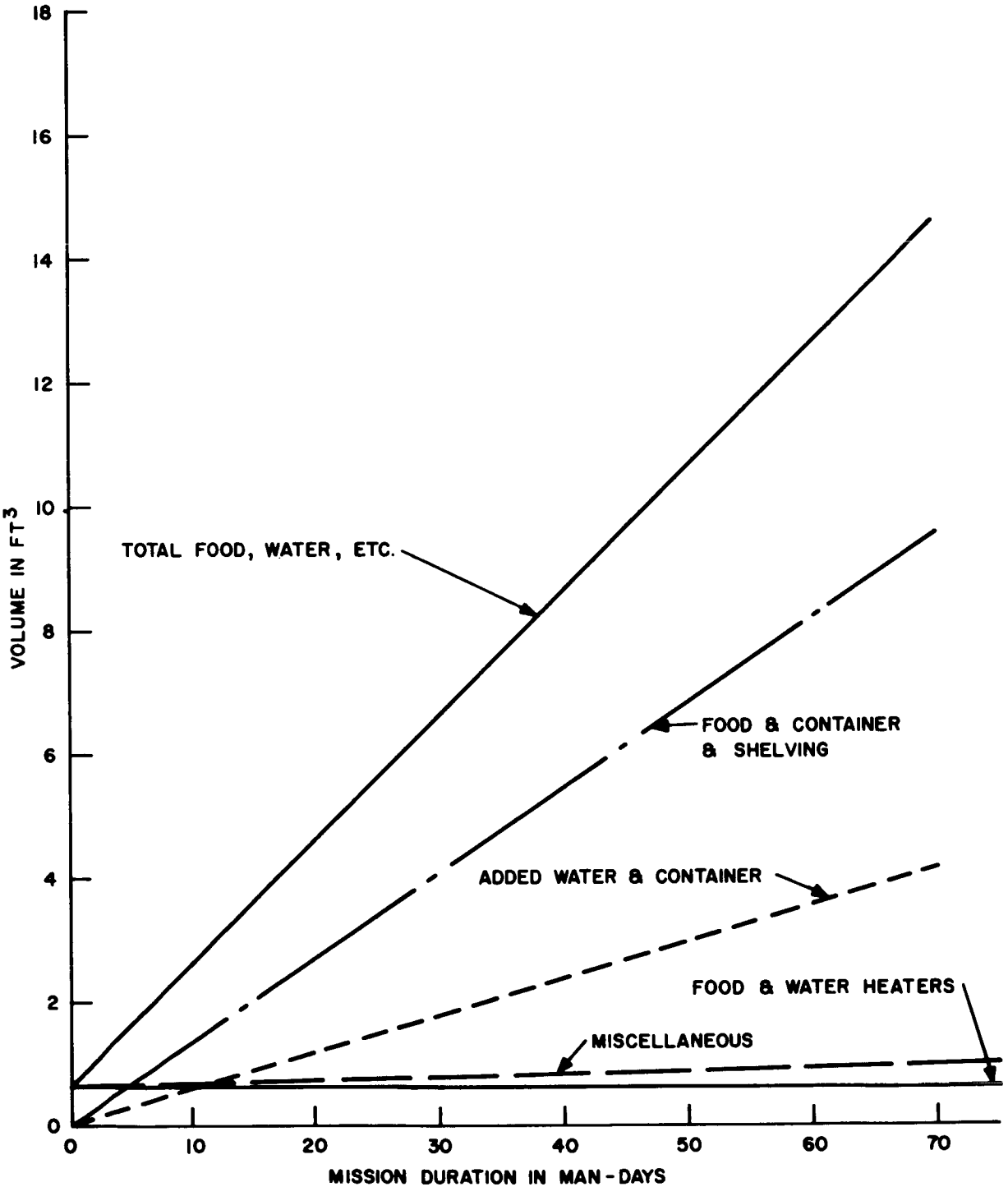


Figure XI-4-6. Nutrient volume versus mission duration — intermediate acceptability menu (3300 kilo-cal/man-day)

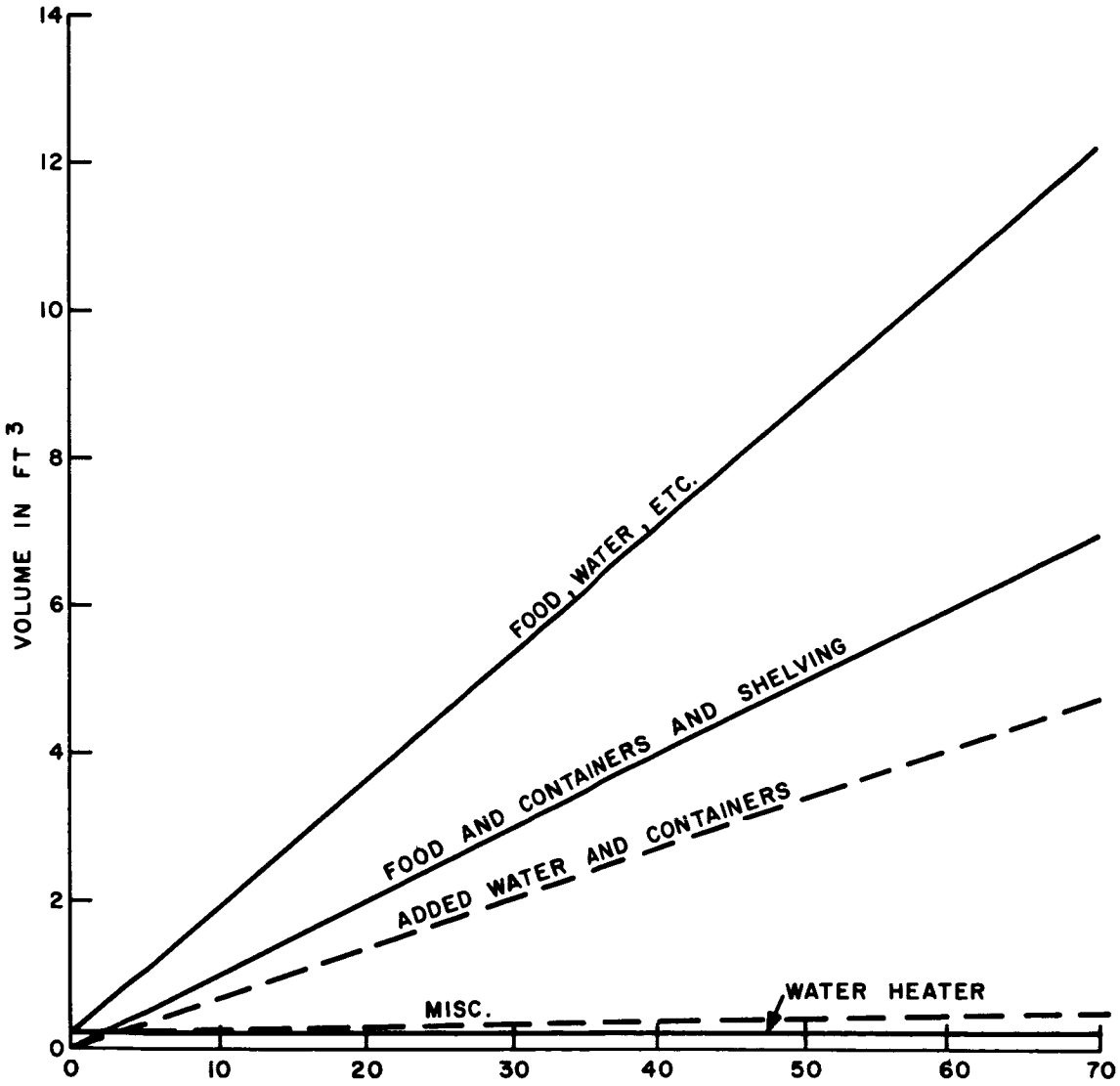


Figure XI-4-7. Nutrient volume versus mission duration — minimum acceptability menu (2500 kilo-cal/man-day)

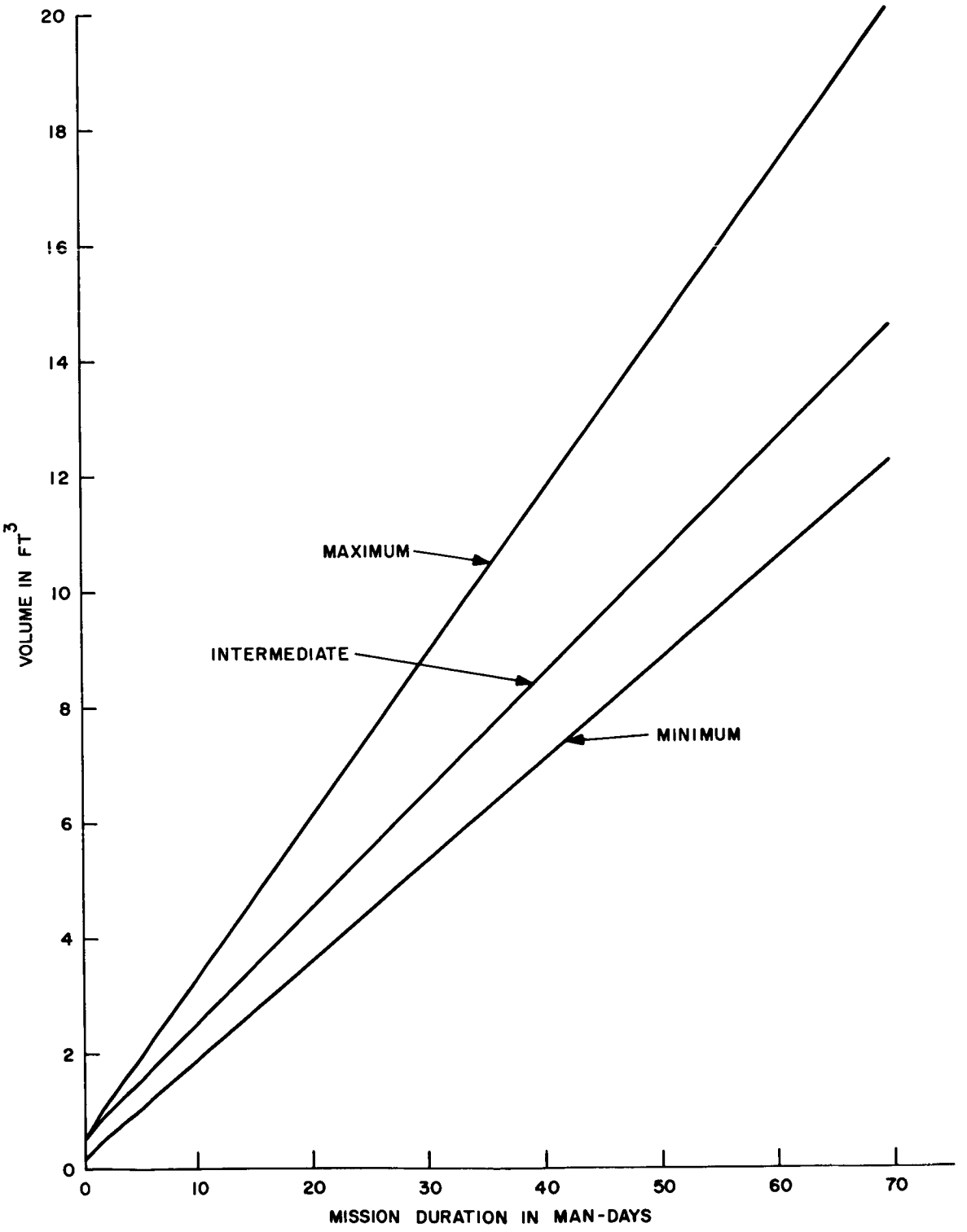


Figure XI-4-8. Nutrient volume versus mission duration — maximum, intermediate, and minimum menu acceptabilities

Figures XI-4-9 and XI-4-10, respectively, illustrate the grand total weight and volume reduced by the amount of water recovered from the dehumidifying system (1.9 pounds per man-day) which can be used to provide a part of the required water.

In summary, the nutrient graphs establish the following weights and volumes for the 3-man—14-day APOLLO mission.

Type Diet	Max.	Int.	Min.
Weight	450	365	310
Difference	85	55#	
Weight less 1.9#/n.d. H_2O recovered from dehumidifier	360	275	220
Difference	85	55#	
Volume	12.3	9	7.4
Difference	3.3 cu ft	1.6 cu ft	
Volume less 50 in ^{3}/md H_2O recovered from dehumidifier	11.1	7.8	6.2
Difference	3.3 cu ft	1.6 cu ft	

A diet of the intermediate acceptability level is recommended for the APOLLO mission. The duration of 14 days is too long a time period to subject the man to the minimum acceptability diet. It is expected that morale would suffer severely and inadequate nutrition resulting from food rejection would also produce a deleterious effect. On the other hand, the mission duration is not considered long enough to warrant the maximum acceptability diet and its inherent penalties in weight volume and power. Thus, the intermediate acceptability diet has been selected.

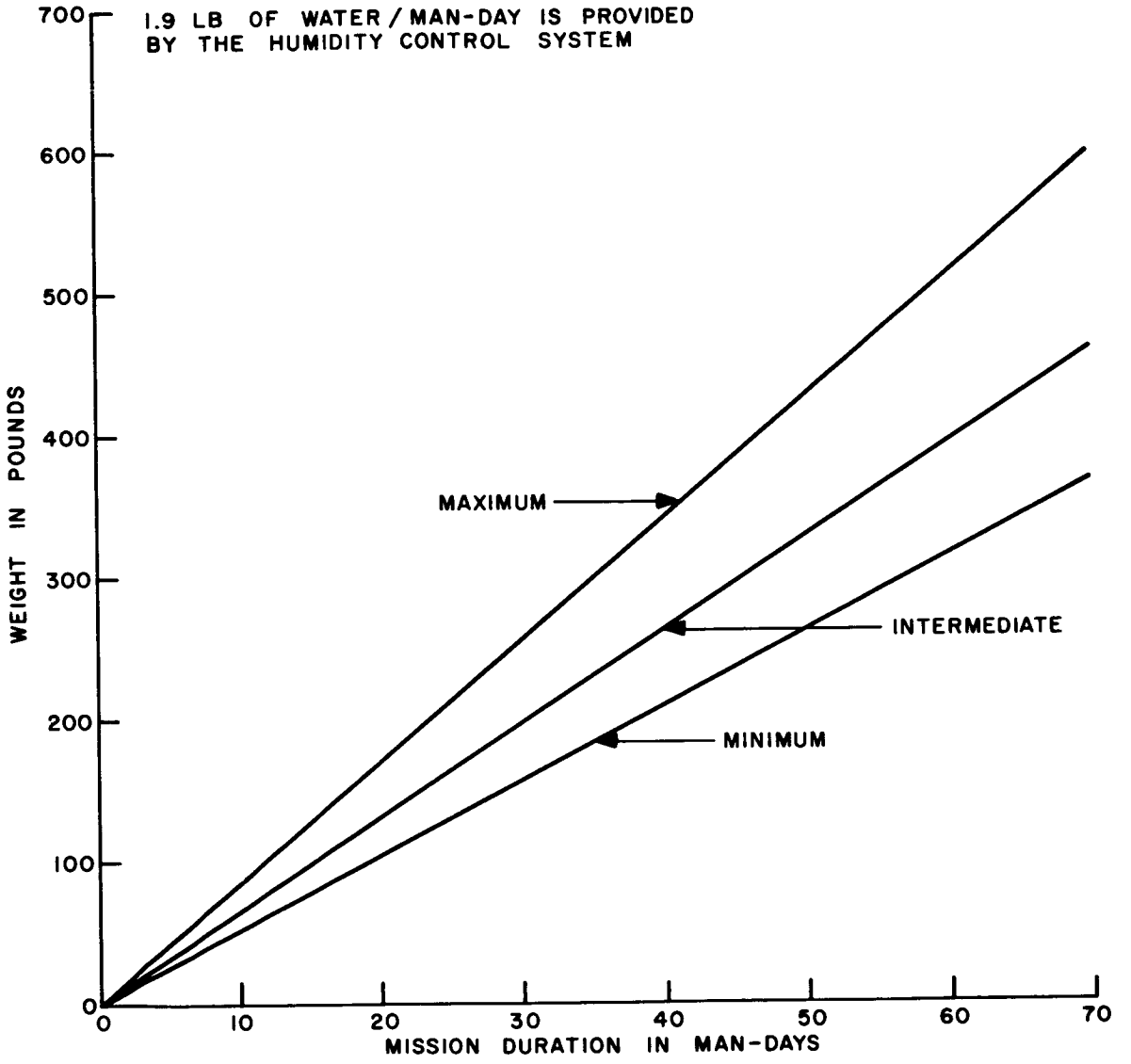


Figure XI-4-9. Nutrient weight versus mission duration — maximum, intermediate, and minimum menu acceptabilities

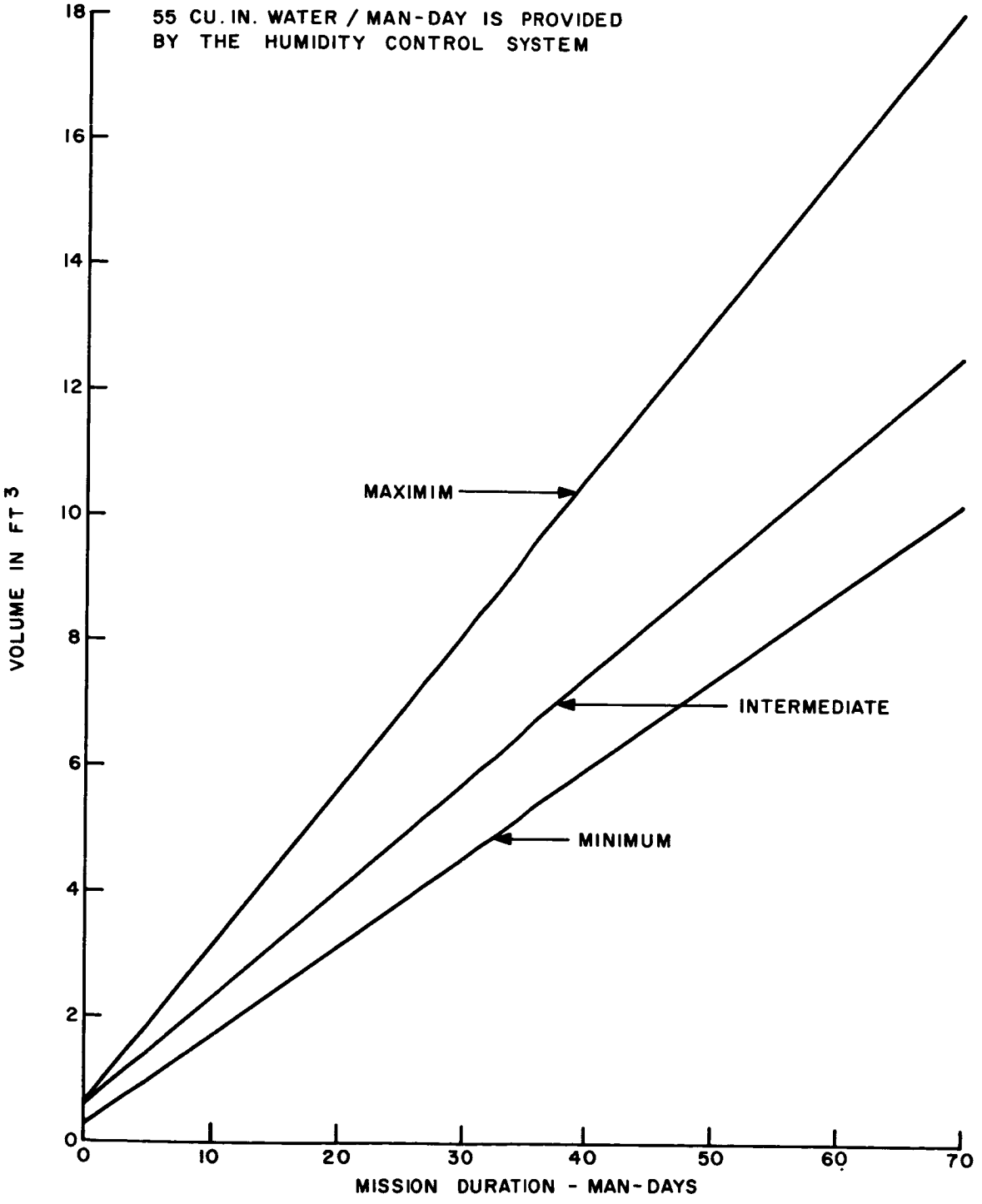


Figure XI-4-10. Nutrient volume versus mission duration — maximum, intermediate, and minimum menu acceptabilities

It is pointed out that the extra 55 pounds which the intermediate diet costs as compared to the minimum menu, represents slightly more than one pound per man-day additional vehicle weight. This small increment appears to be justified when compared with the expected improvement in morale and performance which literature indicates accompanies this change in menu.

WATER SUPPLY

The minimum daily water turnover expected for the APOLLO vehicle has been calculated and presented in the section on physiology. Since the diet proposed requires a slightly larger water input than the minimum diet, the water balance will shift to accommodate this larger turnover.

Based upon the intermediate acceptability diet above, a daily water balance is illustrated.

Input		
	Consumption (food & drink)	5.00 lbs
	Water of food oxidation	0.66 lb
	Total	5.66 lbs
Output		
	Urine	3.43 lbs
	Skin and lungs	1.90 lbs
	Feces	0.33 lb
	Total	5.66 lbs

As discussed in the preceding section, the water content of the food can be assumed to be 1.64 lbs/man/day. This means that supplemental water in the amount of 3.36 lbs is required for consumption by each man each day.

Obviously, this 140 pounds of water could be carried aboard the vehicle as an open supply. However, the relatively large weight has dictated an examination of possible water recovery techniques suitable for this vehicle.

Upon examining water recovery, it is noted that several methods for performing this operation have been developed to the point of demonstrated feasibility. The most attractive of these methods have been analyzed and weight versus mission duration estimates have been made. These estimates are based on a power-weight penalty of 350 lbs./kilowatt of energy used and are illustrated on Figure XI-4-11, which shows the cost of recovering one pound of water per day for each of the methods investigated.

As this curve points out, the most efficient method of recovery is simply the use of the skin and lungs water output which is condensed upon the cabin heat exchanger as excess water vapor. Already a distillate, this water is considered potable and readily acceptable for consumption. Experimental evidence of this is offered through the efforts of the crew of the submarine Skipjack who drank for three days water condensed upon the boat's heat exchangers with no ill effects reported. A coating applied to the heat exchanger to reduce metallic concentration and an activated charcoal bed filter are proposed to further improve the quality and acceptability of this water.

Equipment requirements which include a collecting sponge, squeezing device, filter and small storage tank are estimated to weigh no more than four pounds. From the water balance above, it is seen that 1.9 pounds of the supplemental daily consumption requirement of 3.36 pounds can be obtained in this fashion. The additional 1.46 pounds per man day or 60 pounds per mission will be carried as open supply and will provide the quantity necessary to absorb peak loads until the system stabilizes. Using ten percent storage factor and adding the four pounds estimated for recovery equipment, a total water supply system weight of 70 pounds is estimated for this vehicle.

In summary, total weight and volume required to supply the crew with the intermediate diet is as follows:

Component	Weight (lbs)	Volume (cu ft)
Food	130	5.
Water	70	0.95
Food and water heaters	10	0.6
Containers and shelving	70	1.05
Miscellaneous	5	0.2
Totals	285 lbs	7.80

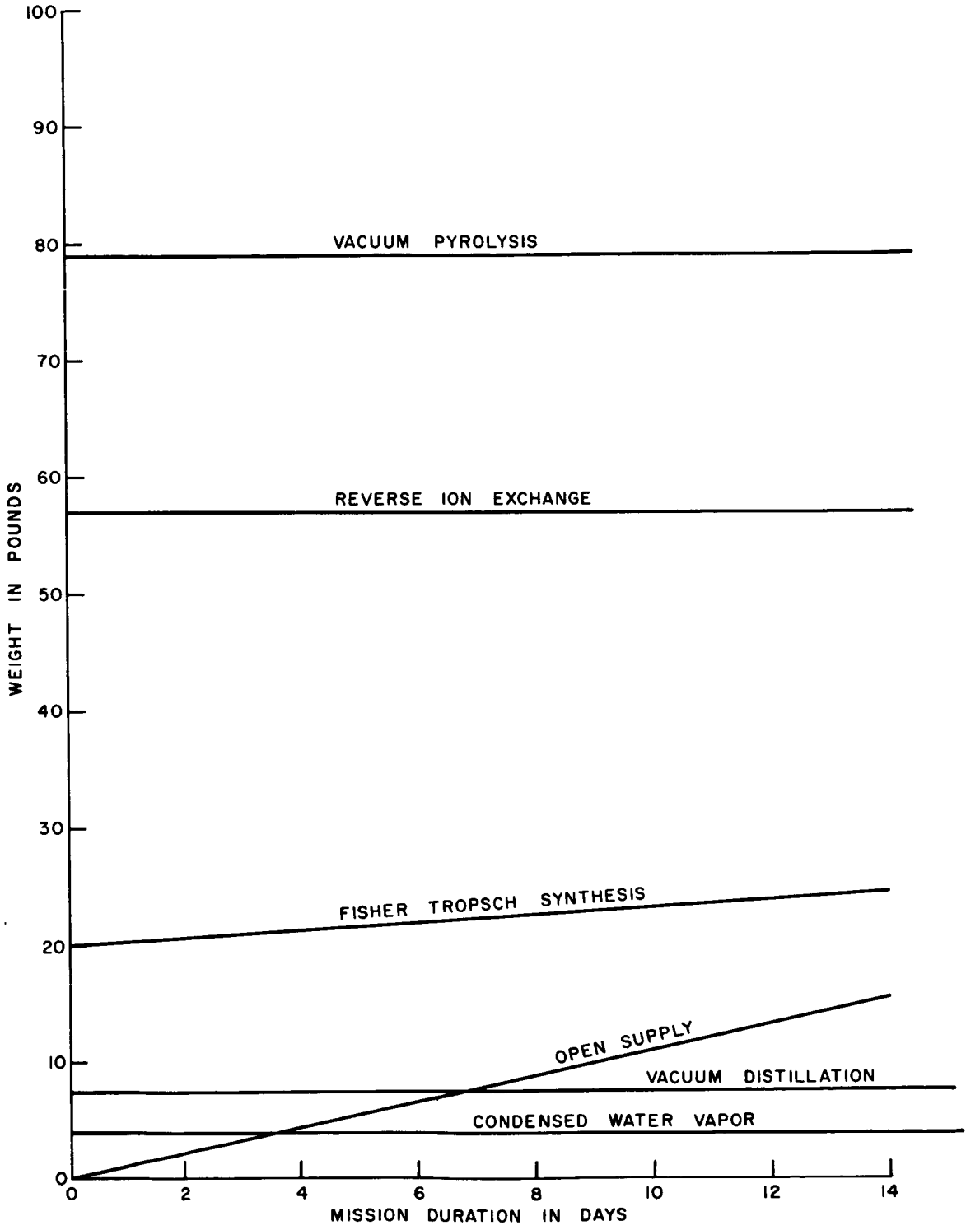


Figure XI-4-11. Total weight required to supply one pound of water per day versus mission duration

5. WASTE

In order to investigate and choose an optimum system design the design criteria for that system must be established. The criteria for a waste disposal system for APOLLO consists of the following requirements:

(1) Mission duration is three men — 14 days

(2) Minimum weight and volume, and no power required

(3) Working parts non-corrosive

(4) Unit to be designed for male use

(5) Urine collection to be used in conjunction with solid collection, as well as separately.

(6) Easily disassembled to be cleaned

(7) Designed to operate in 0.5 to 1 atmosphere pressure and weightless condition.

(8) Odor and bacteria problem eliminated

SOLID COLLECTION

Several techniques for collecting solid wastes were investigated. Among these techniques were materials to absorb feces, such as magnetized absorbant clay; a collector inserted into the anus; feces to be dried by copper sulfate or by boiling moisture off to space and then collecting by an electrostatic precipitator and pneumatic feces-collection system.

The pneumatic feces-collection system was selected as the optimum design. There are several variations of the pneumatic system. One utilizes space as a method of obtaining a partial vacuum inside the collector. This partial vacuum seals the individual and provides a means by which air can be used to guide, by aerodynamic drag, the feces into the collector rather than have it float unrestrained. Another system

utilizes a hand pump as a method of obtaining the same partial vacuum. This was selected as the best design of the optimum technique. The systems' operation is as follows:

The man removes the seat and container caps, unrolls the container, placing the ring with the container overlaying it, and then inserts the assembly in the seat. During insertion, the orientation of the ring is maintained so that the male end of the pressure line pierces the container and enters the female receptacle in the ring. The man straps himself into the seat with his anus positioned over the container and operates the hand pump. A partial vacuum is created in the container by exhausting excess air into the cabin through the relief valve, thus sealing the man's perineum to the container form. The excess air is sucked from the container through the tube, vacuum line, and deodorizer and pumped to the relief valve. The man is now prepared to defecate. (See Figure XI-5-1.)

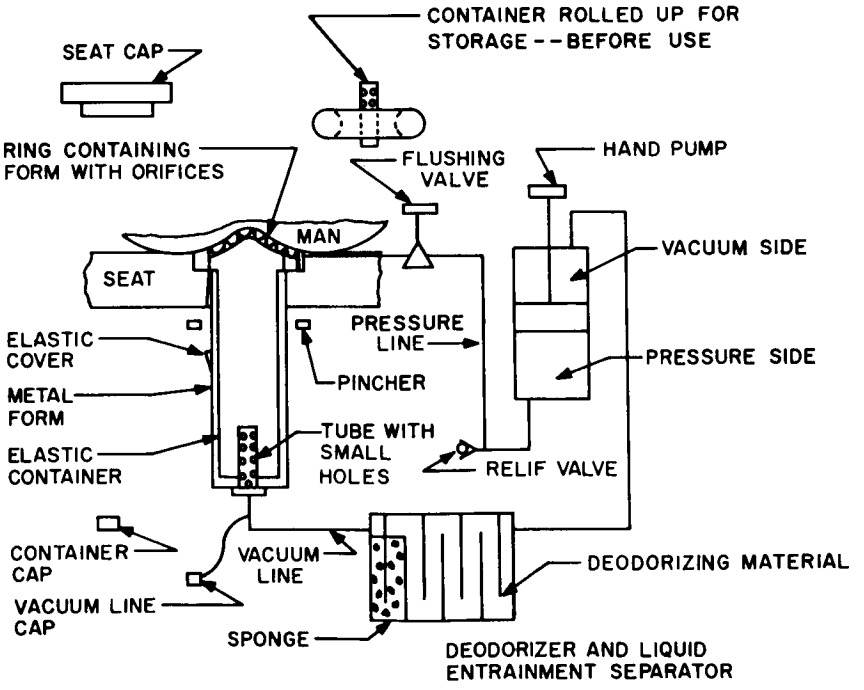


Figure XI-5-1. Pneumatic feces collection system

During defecation, the pump is operated as required to maintain the vacuum. When defecation has been completed, the flushing valve is opened and the hand pump operated. Air and gases from the container circulate through the tube, vacuum line, deodorizer, pump, pressure line, from and back into the container through orifices spaced around the container ring. Pumping is continued until the odors have been removed and the jets of air from the small orifices in the ring have forced the feces deep into the container.

The pinchers are then operated, squeezing the container, and sealing the feces in the bottom. The man removes himself, wipes his anal perimeter, and places the paper in the container. The ring is removed from the container and the container top is closed with a spring clip. The vacuum line is disconnected and it and the container are capped. The container is removed from the seat and the seat capped to complete the operation.

This system has the advantage of not requiring the venting of cabin gas to outer space and has a container which can be cleaned and re-used. Sanitary conditions are easily maintained by the fact that the crew member's perineum is separated from the collection seat by the disposable container and feces never contact the seat or system.

The associated mechanical and chemical equipment is light in weight and easily stored under a minimum size seat. The feces collection container is small in volume in comparison to other systems' containers, since flatus and flushing air is deodorized and vented to the cabin. This system, with its hand pump, gives the operator complete control over the system and no electrical power is required.

URINE COLLECTION

The collection of urine in the absence of gravity is not as difficult as the collection of feces, since urine is expelled from the penis with an imparted velocity.

The receptacle for the penis is a self-sealing sick-room device. It is formed of rubber and has a receptacle containing a diaphragm to prevent backflow of urine while the patient is lying in a supine position. This seal technique is reliable and can be used under weightless conditions. The container which collects and retains the urine consists

of a bellows-type collapsible container. To operate, the bellows is squeezed together and the penis is inserted into the attached receptacle. The crew member then micturates and simultaneously releases the bellows at a rate which still provides a slight vacuum.

A three-way spool valve at the neck of the container allows the release of the penis by bleeding off the partial vacuum when micturation is completed. The valve can then be pushed to the closed position.

Should it be desired to reprocess the urine, there must be some provision made to physically transfer the urine from the container to the water recovery system. To unload the container, the neck is inserted into the processing device, the valve pushed to open position, and the bellows squeezed to force the urine from the container into this water recovery system.

CONCLUSIONS

It was found that the most desirable method of collecting feces is a hand-operated pneumatic system utilizing a disposable container. A re-usable type of container could be substituted for the disposable container without changing the unit design. By venting to outer space, the hand pump could be eliminated but the loss of gas would have to be made up from a storage supply.

It was determined that the most suitable method of collecting urine is a self-opening, collapsible container attached to a modified sickroom-type penis receptacle. The collapsible container was found to have the advantage of not requiring an external source of evacuating power. Also, should it be desired to reprocess the urine, it could easily be transferred from the container to the processing equipment.

6. SURVIVAL EQUIPMENT

Equipment suitable to support the life of the APOLLO crew for 72 hours after an unprogrammed landing will be provided within the re-entry vehicle. This equipment will provide sustenance and protection, assuming either water or land impact at any spot or in any environment on Earth.

Maximum use of the vehicle can be expected should a ground recovery be made. That is, the capsule can be used as shelter in particularly cold climates or the parachute used as cover in desert-type environments. Although the capsule is designed to float, a life raft is provided in the event that damage results upon landing and the vehicle ships water. The raft will be packaged, if possible, in the parachute housing so that it is readily available after emergency egress. Consideration shall be given to automatic jettisoning and inflating of this raft at 'chute release.

Except for those particular items (for example, the raft and the equipment normally stowed within it) which pertain to water survival only, the equipment will be packaged in readily transportable containers, such as knapsacks, in the event that the decision to "walk out" is made.

A preliminary list of the survival equipment proposed for this vehicle is presented in Table XI-6-1.

TABLE XI-6-1. SURVIVAL EQUIPMENT

ITEM	WEIGHT (LBS)	VOLUME (CU IN)
Chap stick	0.03	1.
Insect repellant (3 required)	0.93	24.
Snake-bite kit	0.23	9.
Sunburn ointment, can (3 required)	0.48	15.
Fishing kit	0.47	15.
Matches, 170/roll	0.50	36.
Life raft, 3-man	33.00	4000.
Compass	0.32	44.
Manual, survival	0.44	15.
Salt tablets	0.09	1.
Universal clothing kit (3 required)	14.25	1050.
Mirror, signal	0.36	8.
Smoke, signal (3 required)	1.35	24.
Individual food packet (9 required)	19.08	360.
Axe, gobal	1.69	238.
Knife, hunting	0.28	20.
TOTALS	73.49	5860.

The problem of supplying water for survival is particularly difficult since the most severe case to which one would design is that of desert impact where water consumption requirements are high. The best available information indicates that between six and eight pounds of water are required per man per day in this environment, assuming either no travel or travel at night only. Since the urine output under these conditions is expected to be somewhat less than one-half the consumption requirement, supplementary water must be provided even if urine recovery could be effected.

This problem is particularly real, not only because of the large weight involved, but also because of the possible choice of the Woomera Range in Australia as a secondary landing site. For these reasons, an analysis of methods of recovering water in the vehicle during flight and storing it for emergency use has been initiated.

Preliminary results of this study have indicated that, since an aborted flight could be as short as one day, sufficient urine could not be collected and recovered for total satisfaction of the survival requirements. One day's urine output is estimated at four pounds. If we compare this with the 35 to 40 pounds required to recover this water by vacuum distillation, it is immediately apparent that the cost of recovery is greater than the product recovered.

As a result, continuing study will be applied to the recovery of urine after landing and consequent reduction of open supply requirements. Another area which will be investigated is that of combination of the residual oxygen supply with hydrogen to produce water.

7. LIGHTING

In consideration of the tasks to be performed, the crew will have sufficient illumination to read placards, check lists, manuals, maps, instruments, etc. In addition, lighting is provided so that the crew can become dark-adapted for interpretation of the CRT displays. For use in the vibration and shock environment arising during launch and re-entry, a new type of lighting is introduced.

REQUIREMENTS

Specification MIL-L-6503A (USAF) describes the light intensity requirements for various locations in military aircraft. The applicable sections of this specification have been used as the guide for selecting the lighting system for the vehicle. Light requirements, per the military specification, are listed in Table XI-7-1. These lighting requirements were applied to the vehicle for lighting the crew station, console lighting of edge-lit panels, warning lights, passageway lights, reading lamps, worktable lights, and mission module area lighting.

TABLE XI-7-1. LIGHT REQUIREMENTS

Location	Light Intensity Foot-Candles		Color	Dimming Required
	Min.	Max.		
Passageways (on floor) where dark adaption need not be maintained.	12.	20.	White	NO
Passageways (on floor) where dark adaptation must be maintained.	0.1	0.5	Red	NO
Crew station locations (on working areas) where map reading, course plotting, etc. is accomplished.	8.	15.	White	YES

Task and Visibility

Although MIL-L-6503A specifies 15 footcandles maximum for crew stations, studies by Dr. H. Richard Blackwell indicate that this intensity would be equivalent to an overcast sky and would be insufficient for reading small, indistinct print such as might be found on maps, charts, or manuals. Therefore, a design minimum of 50 footcandles was chosen as desirable. This increase of light would also increase the accuracy, speed, contrast sensitivity, visual acuity, and convergence reserve of reading.

In attempting to comply with or exceed the above requirements, three types of lighting systems were investigated—incandescent, fluorescent, and electroluminescent. The physical and electrical characteristics of each of the three systems will be compared and a final system configuration recommended.

ILLUMINATION SOURCES

Three sources of light are considered in this study. A point source (incandescent light bulb), a line source (fluorescent), and an area source (electroluminescent). A comparison is made of each type of lighting with respect to lighting function, such as crew station illumination, panel lighting, warning lights, utility light, passageway light, reading lights, worktable light, and emergency light.

Incandescent Lamp

For crew station lighting with incandescent lamps, several fixtures (such as AN 3400-3) could be used. This particular fixture contains both red and white illumination in one assembly. The lamps normally used in the fixture are MS 25235-SB 311 for clear illumination and MS 25232-R307 for red-light adaption. For improved shock and vibration resistance, the lamps proposed for use in this fixture are the GE1680, six-volt bulb and the GE1723 bulb.

Red light is provided for the dark-adaption needed for efficient utilization of CRT displays. A separate fixture will be provided for each crew station. One integral

switch-rheostat is provided for clear illumination and one rheostat for red light illumination. The rheostats will provide dimming from 100 percent brightness to one percent brightness and include an integral on-off switch.

The maximum power requirements for the clear illumination is estimated at 108 watts. For red illumination, the maximum wattage is estimated at approximately 56 watts.

The total weight of the incandescent system, including two fixtures, bulbs, and two rheostats is approximately two pounds.

Lamp life is an important consideration. The 1680 and 1723 lamps are rated for 300 hours life. However, by use of rheostats, the lamp life can be increased tremendously by reducing the voltage across the lamp. Data supplied by General Electric Lamp Department at Nela Park show that decreasing the voltage to 80 percent of rated value increases life from 300 to 3000 hours and 50 percent candlepower still would be maintained. Therefore, for a long mission life, incandescent lamps would be feasible if operated below rated voltage.

The environmental conditions to which the lamps would be subjected is also important. Although the applicable lamp specifications, WL-111C and MIL-L-6363, do not specify environmental tests such as shock, vibration, acceleration and temperature, the lamps must, nevertheless, perform under environments which include these factors. Tests have indicated that incandescent lamps are subject to failure when exposed to these environmental factors at the magnitudes expected during flight. Those tests have shown, however, that a cold filament is able to withstand the shock and vibration environment that is expected.

Fluorescent

The main consideration favoring the use of fluorescent lamps is the higher efficiency obtainable. A value of approximately 50 lumens per watt is obtainable in fluorescent lamps, while 15 lumens per watt is an approximate efficiency figure for an incandescent lamp. In order to obtain a maximum 50 foot-candles on the subject, the following calculation were made. With a space ratio of four, the coefficient of utilization would be 0.47. Assuming a maintenance factor of 0.7 and using a 15-watt lamp of 750

lumens, five lamps would be required. However, due to space limitations, four lamps will be used, two per fixture. No red-light adaption is provided.

In addition to the large space requirements of fluorescent fixtures, total weights would be higher. The weight of the ballast, if 115-v 400 cps power is used, is estimated at two pounds per fixture. In addition, a dimming ballast would weigh an additional pound. The weight per fixture would be about eight pounds including the ballast or, for two fixtures, a total of 16 pounds. Since breakage of the lamps is possible, enclosed types of fixtures must be used.

Fluorescent lamp life, however, is very much longer than that of incandescent units. For the F15T8 lamp, 7500 hours of lamp life can be expected at full brightness. The lamps, themselves, would be able to withstand the environmental shock, vibration and acceleration requirements. However, light output of fluorescent lamps is greatly affected by the temperature of the surrounding air. Extremes of heat and cold cause marked reduction in the amount of light produced. Special ballasts must be designed for operation below 50 F. Lamps must be enclosed at low temperatures or output could drop to about 20 percent at zero F. Power requirement for the fluorescent system is 90 watts. In addition to the weight of the fixture, conversion equipment (such as an inverter) would be required to convert the d-c battery power to 115V 400 cps a-c power for the fluorescent lights. The weight of a converter is approximately five lbs.

Electroluminescent

Electroluminescence is an area source of light. Electroluminescent panels resemble a sealed parallel plate capacitor in structure. One of the plates must be a transparent conductor to allow the passage of light. The dielectric is an insulator in which the phosphor (usually copper-activated zinc sulphide crystals) is suspended. The unit is sealed with two sheets of laminated Kel-F plastic. Light is emitted when the lamp is actuated with low-frequency alternating current.

A decrease of brightness with time is the most serious problem with electroluminescence. The curve resembles an exponential decay. Brightness-versus-time curves for

160-volt sine wave and 120-volt sine wave power are shown in Figures XI-7-1 and XI-7-2. These curves suggest that this type of lighting be used for limited background only. As can be seen by the curves, more or less of a "plateau" is reached after the light has decreased to half brightness. While not providing a constant light source, the light could be maintained above 10 foot-candles for more than 210 hours. While not sufficient for detailed reading tasks, this light provides a very pleasant and calm source of illumination. The panels are less than 1/16 inch thick and can readily be adapted to mounting on the walls of a space vehicle.

The power requirements for electroluminescent panels are about 0.15 watt per square inch. To give sufficient background illumination of 10 foot-candles, three panels 10 by 11 inches would be used. Power requirements, therefore, would be about 50 watts.

The weight of electroluminescent panels is one of their main advantages. Three panels of the size mentioned would weigh about 0.6 lb. An inverter would be required; this would add an additional two lbs to the system. Electroluminescent panels should perform well for the levels of environmental vibration, shock, and acceleration expected, because of their extreme light weight and thinness and because they are, in effect, a solid-state device.

Temperature data shows that, as ambient temperature increases, light output increases. At 140 F, the brightness is increased to 120 percent of the brightness at 80 F. However, as temperature is decreased to 40 F, light output is also decreased to 70 percent, the brightness at 80 F.

While not considered for flood-type lighting, electroluminescent panels are ideal for background lighting. They are light in weight, can easily be dimmed, and are adaptable to the configuration.

CREW STATION SYSTEM DESIGN

Compartment Lighting

The lighting system proposed for the crew station is a combination of incandescent and electroluminescent systems. Electroluminescent lighting would be used at lift-off

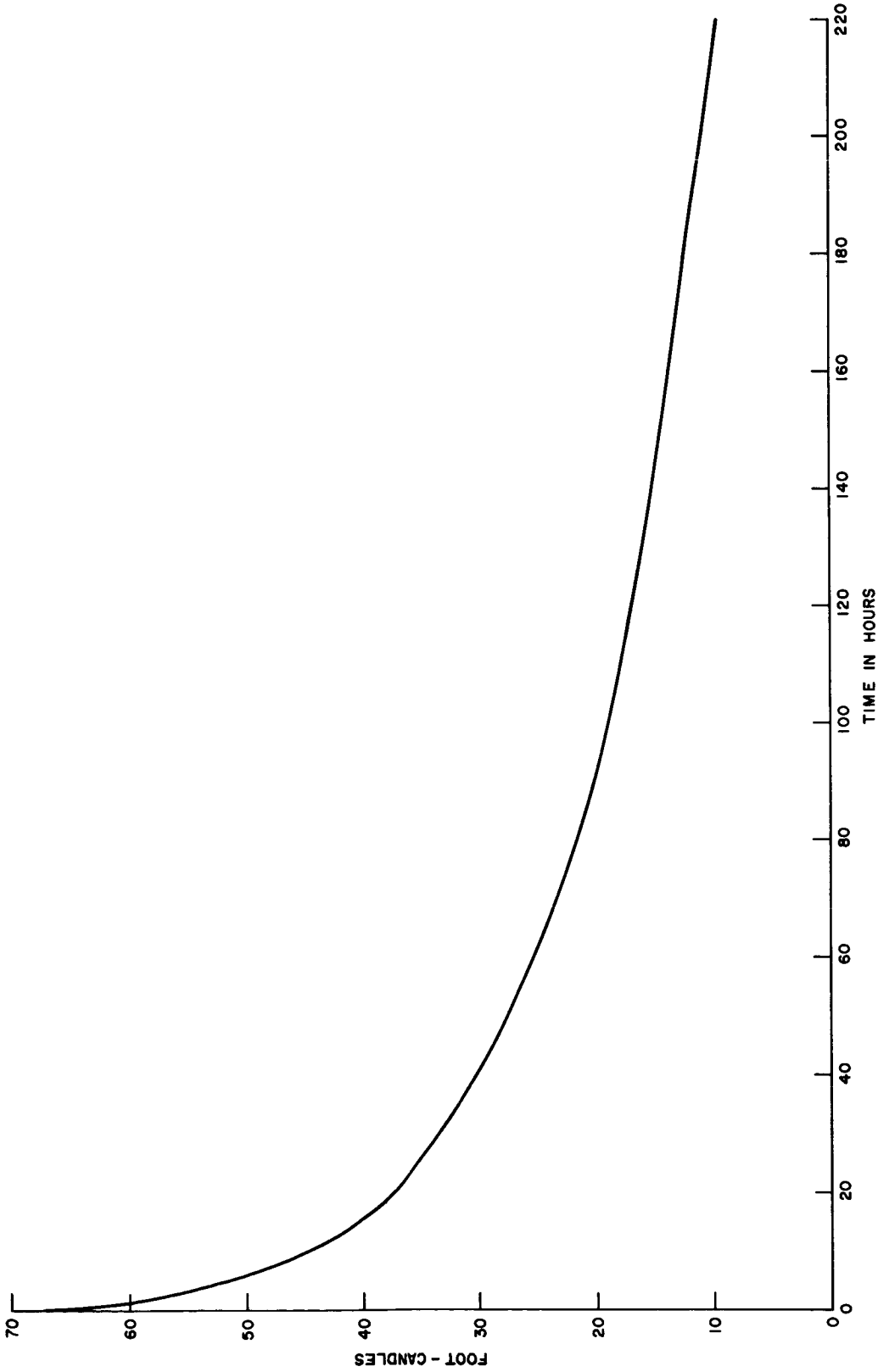
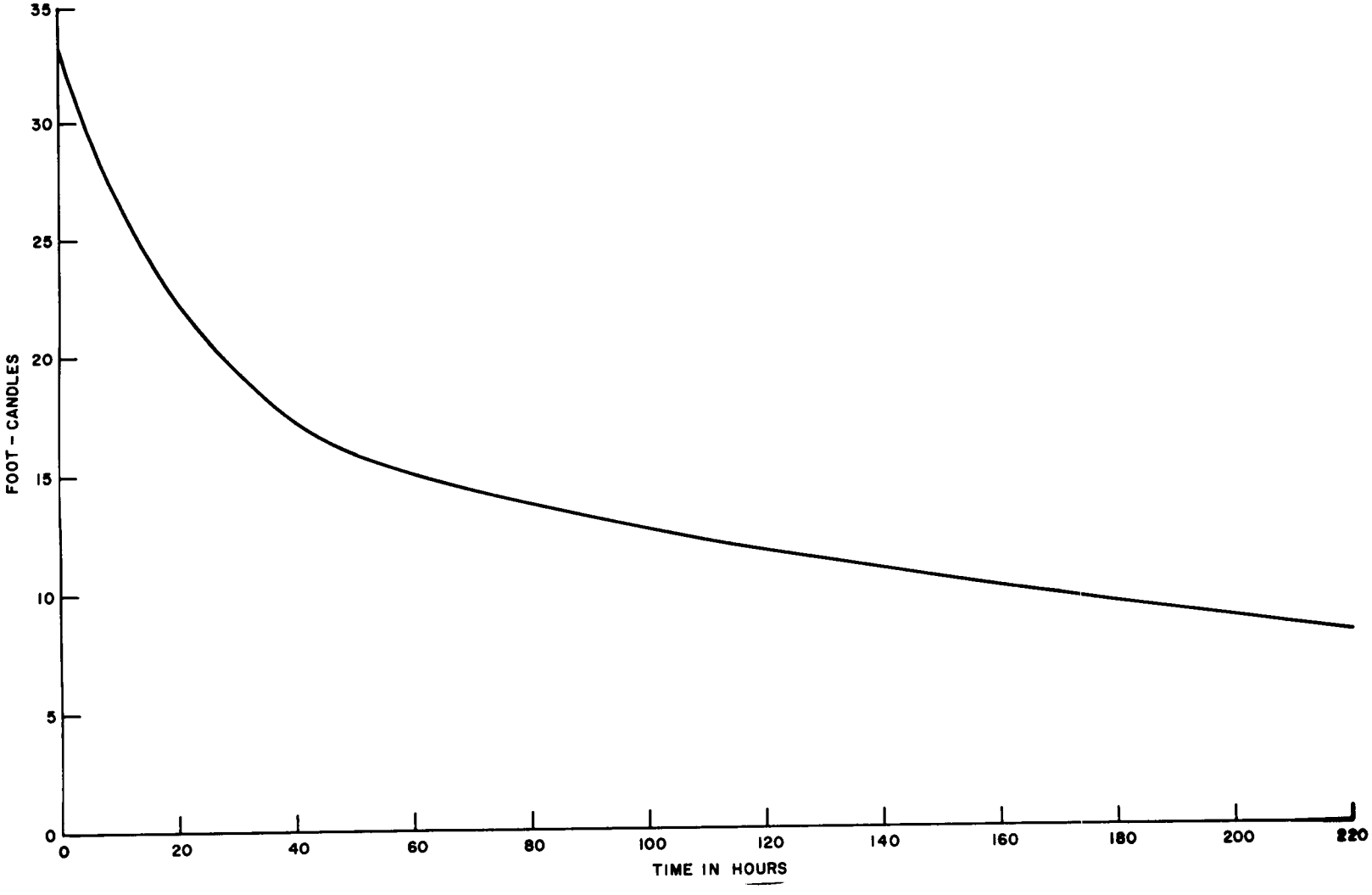


Figure XI-7-1. Brightness versus time for General Electric electroluminescent panel, green, 160 V, RMS, sine wave cps

CONFIDENTIAL



CONFIDENTIAL

Figure XI-7-2. Brightness versus time for General Electric electroluminescent panel, green, 120 V, RMS, sine wave cps

when vibration would be present, and at re-entry and parachute opening when shock loads are a factor. Since incandescent lamps with cold filaments are much more resistant to shock and vibration, this difficult environmental problem is solved by the intermittent use of the electroluminescent panels. Therefore, the incandescent lamps would not be used during the orbit condition. The weight of the proposed system is only 2.6 lbs. As a comparison, the weight of the fluorescent system would be 18 lbs.

Table XI-7-2 shows a comparison of the various types of lighting considered for the crew station.

Console Lighting

Edge-lit panels and integrally lit instruments will be combined to provide good instrument panel and console illumination. As mentioned above, dark-adaption is required when integrated use of scopes and other displays is necessary. Therefore, red panel light will be provided and designed to produce a balanced illumination level with an absolute minimum of hot spots.

It is estimated that 24 NE-2D neon bulbs will be required at a total power requirement of 2.4 watts. The integral lighting requirements for the displays and read-outs is assumed to draw an equivalent amount for a sum total of five watts to completely light the panels.

Emergency Lighting

Emergency lighting (per Mil-L-6503A) will be provided and shall consist of a battery-operated flashlight installed near each crew member seat. The mounting will securely hold the flashlight in place but permit easy removal in case of emergency. A decal, illuminated in the dark, will be placed on the mounting. The flashlight will be explosion-proof, watertight, rubber-encased.

Crew Station System Design Considerations

One of the prime considerations for space travel is weight optimization. In order to keep power requirements to a minimum, interlocking switches will be provided so that all lighting will not be on simultaneously. Of course, panel light and warning

TABLE XI-7-2. COMPARISON OF ILLUMINATION SOURCES FOR CREW STATION LIGHTING

	MIL SPEC APPROVED	MAX ILLUMINATION	INTEGRAL DARK-ADAPTATION	DIMMING CONTROL	LAMP LIFE	WEIGHT	COST	AREA SQ FT	MAXIMUM POWER	ENVIRONMENT	TOXICITY IF BROKEN
		FOOT-CANDLES			HOURS	POUNDS			WATTS		
INCANDESCENT	YES	W 50 R 21	YES	YES	300	2	$40	1.25	GOOD R-56	GOOD	NO
FLUORESCENT	NO	50	NO	YES	7500	18	$2400		90	GOOD	POSSIBLE
ELECTRO-LUMINESCENT	NO	10	NO	YES	300	.6	$2030	2.3	50	GOOD	NO
INCANDESCENT AND ELECTROLUMINESCENT						2.6	$2070	3.55	108	GOOD	NO

CONFIDENTIAL

CONFIDENTIAL

light power will be available continuously. If electroluminescent power is on, crew-station incandescent lighting or crew-member floodlighting will be off. Also, floodlighting will be used for only one crew member at a time to reduce total power required. This limits the maximum power for lighting to 130 watts.

MISSION MODULE AREA LIGHTING

Requirements

Mission module lighting requirements are for a compartment approximately 8 feet long, 2.5 feet wide and 8 feet high. On one side of the compartment will be vehicle flight and mission equipment and on the other side the galley and sleeping area. A short passageway connects the recovery vehicle and mission module.

COMPARTMENT LIGHT

From preliminary design data, it appears that mission information will be presented in the mission module, which may require red light adaptation. Fixture AN 3400-3, which contains both red and white illumination, is suggested. Two fixtures are required for the mission module compartment. The incandescent bulbs would be used for orbit lighting of the compartment and will require a maximum of 72 watts.

LOCAL LIGHTING

Spotlights will be provided in the lavatory, in the galley, and at the flight equipment console. Each fixture will draw 20 watts, but they will be operated intermittently and hence can be assumed to require no more than 30 watts continuously. Edge-lit panels and integrally lit instruments will be used where required on the flight console. Five watts is estimated for these lamps.

PASSAGEWAY

The passageway between the crew station and mission module area will be illuminated with electroluminescent lighting. One two-inch by five-inch panel at each end will provide sufficient light for passage between the two compartments. These panels

have the advantage that their thinness does not interfere with passage and power consumption is low. The power consumed by two two-inch by five-inch panels would be three watts.

FUTURE DESIGN CONSIDERATIONS

The above lighting system for the APOLLO vehicle is based on the best present state-of-the-art lighting available. The power required during the orbit phase of the flight is a maximum of 242 watts. Even though a solar cell battery system is used, additional power requirements increase overall system weight.

Future investigation will seek an improvement in electroluminescent lighting. For only short times can a 40- to 60-footcandle intensity be obtained at the panel surface by careful selection of vendor, color, voltage, and frequency. At present, this illumination can be obtained for only about 15 hours. However, it may be possible to increase brightness for a period of 300 hours by choice of phosphors and panel construction. Investigation to date has shown that the light output under similar conditions varies considerably with various vendors.

Another area of improvement would be the use electroluminescent lighting for consoles. Much work could be done using vari-colored lights to signal flight functions. Although power requirements may be the same as those of neon bulbs, illumination would be greater and the weight-saved ratio would be about 3 to 1.

Use of electroluminescence for the entire mission would reduce the lighting weight by about a 3-to-1 ratio. The saving in power would also be about 3 to 1.

8. RECREATION

It is generally accepted that the crew of a space vehicle will be required to perform some exercise in order to maintain muscle tone. This is believed to be true even when exposure to zero "g" is limited to the relatively short 14-day mission of the APOLLO vehicle.

Estimates of the amount of exercise required vary, but it appears reasonable to assume that a mild daily workout of 10 to 15 minutes duration will be sufficient to prevent muscle-tone decay. Naturally, the exercise specified and the equipment with which it is performed must be of a type which is suitable to the weightless environment. At its simplest, the program could consist of body muscle tensioning effected through opposing muscles. The equipment required here is little more than a length of rope through which the tension is directed. A set of commercially available exercising springs is also offered as an excellent method of achieving the desired results while weightless.

Since man is aboard the vehicle and since he is capable of performing work, it has been suggested that he be employed to produce a useful product as a result of his labors. This has been analyzed and the results presented herein.

If it is assumed that the efforts of the crew members can be enlisted on a regularly scheduled basis to produce vehicle power, then it is conceivable that the over-all power supply weight might be reduced as a result. Furthermore, in order to trade-off the weight of the man-propelled power generation equipment versus the greatest output, a hypothetical work cycle of 15 minutes on and 15 minutes off has been assumed for each of the three crew members for eight hours a day. Under these cyclic conditions, it is expected that a man would produce 250 watts of energy per cycle or an equivalent 125 watts continuous for the eight-hour period. Based on a three man crew, this would provide a continuous effective energy output of 125 watts or 3000 watt-hours per day.

In accomplishing this, however, food, water and oxygen requirements will increase, since the activity level would be significantly higher than that expected under normal

vehicle habitation. In addition, the increased oxygen consumption produces, in turn, an increase in CO_2 production and resultant collection system growth. Figure XI-8-1 indicates the vehicle weight increase estimated for the food, water, oxygen, and CO_2 collection equipment which would be necessary to support this activity. As indicated, the supplementary requirement per man per day is 11.15 pounds for food and water, 3.55 pounds for oxygen and converter and 3 pounds for CO_2 collection equipment.

If an estimated 30 pounds is added for energy conversion equipment and the above figures are extended to the 42 man-day mission limit, 775 pounds of support weight are required to provide 125 watts of continuous power. This is now traded off against the vehicle power system weight of from 300 to 600 pounds per KW, and illustrated in Figure XI-8-2. From this, it is clearly seen that man is a relatively inefficient source of vehicle power and cannot logically be considered for this task in space.

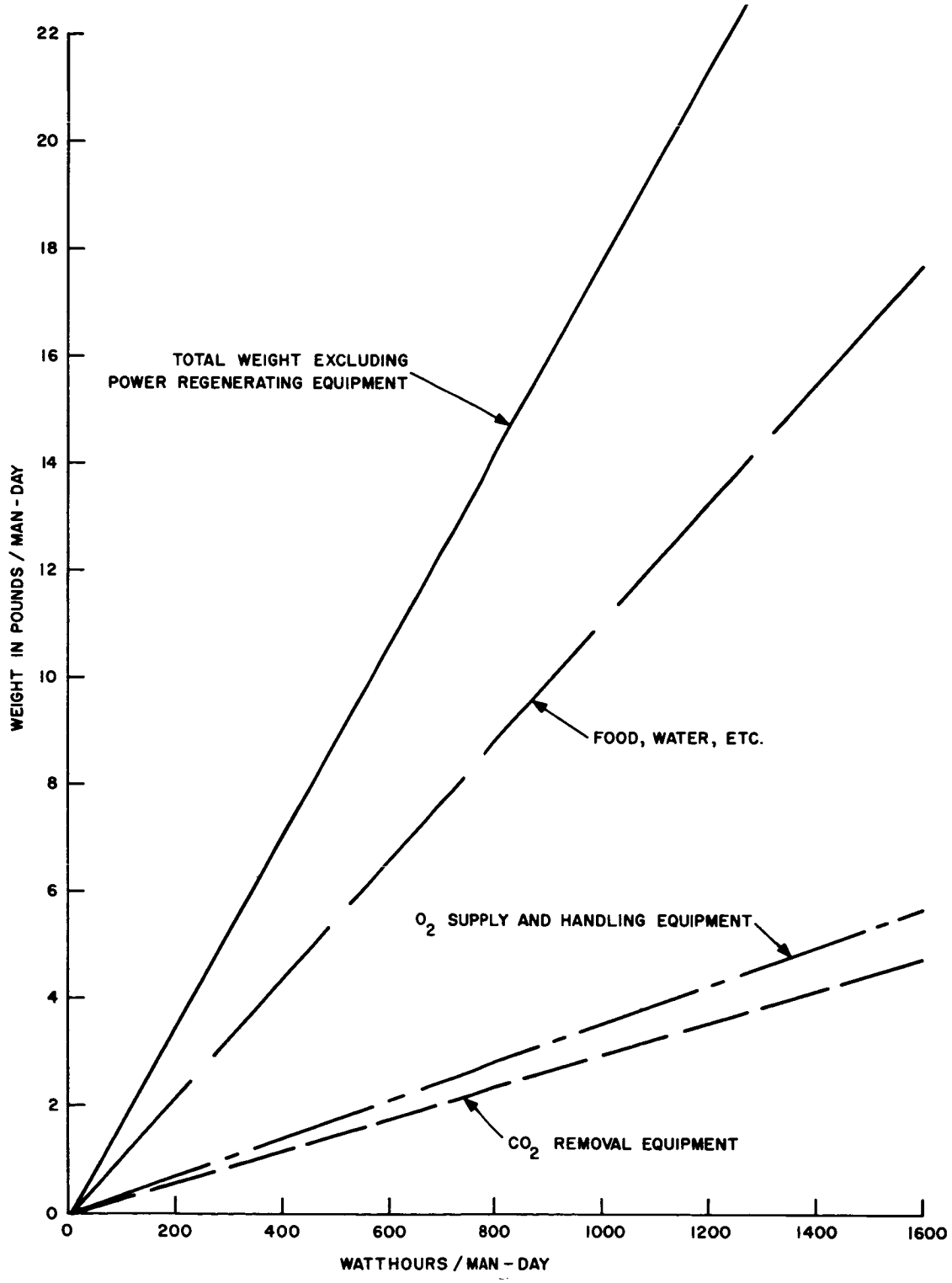


Figure XI-8-1. Vehicle weight increase versus watthours

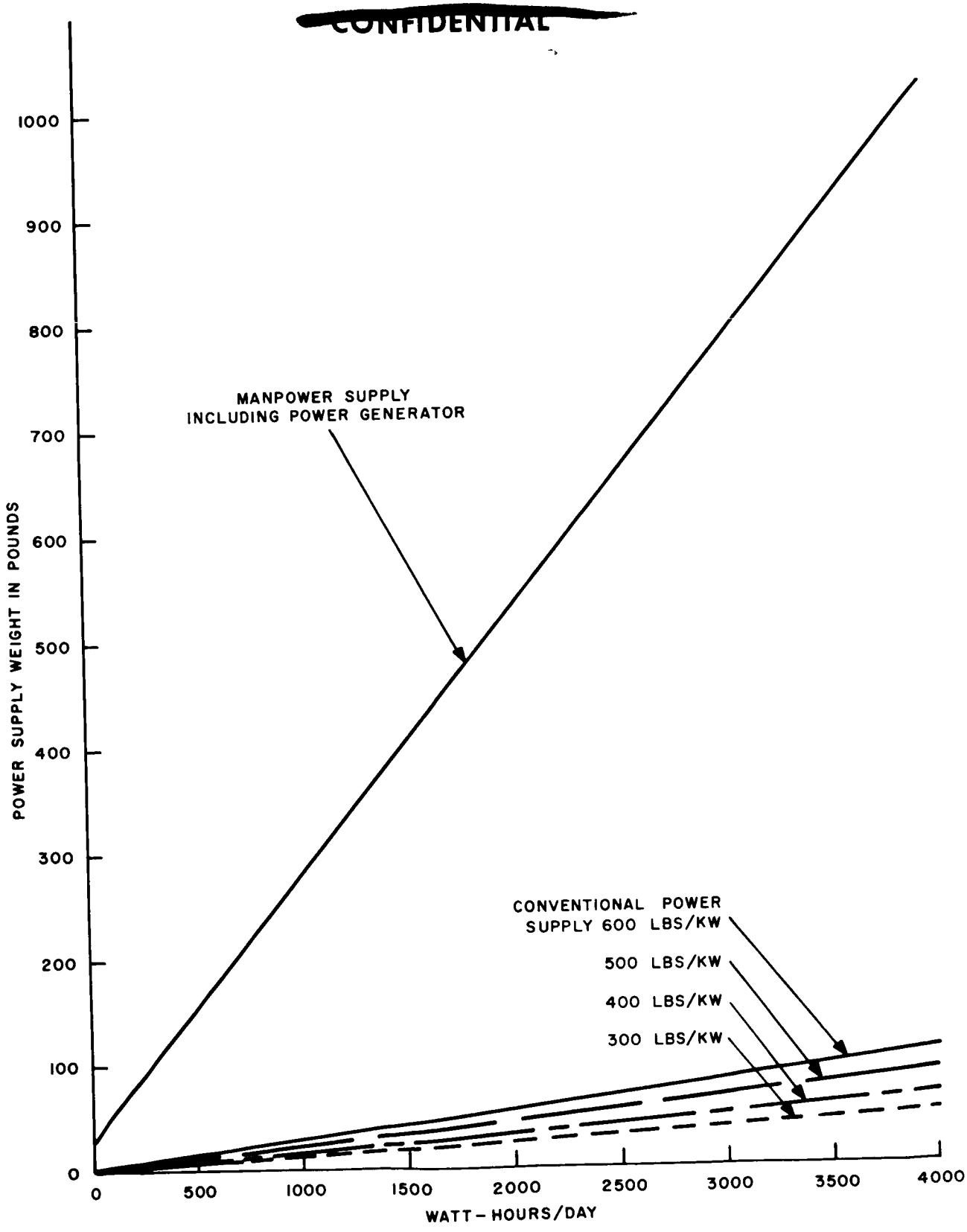


Figure XI-8-2. Weight of power supplies versus watthours/day for 3-man, 14-day mission

9. ACOUSTICAL ENVIRONMENT

The primary sources of acoustic noise which affect any re-entry vehicle may be divided into three categories:

1. That which arises from rocket-engine operation during static firing and missile launching.
2. That caused by the turbulent aerodynamic boundary layer during the vehicle boost phase.
3. That caused by the turbulent aerodynamic re-entry boundary layer and wake of of the re-entry body.

The glide phase following boost may be mentioned briefly here for the sake of completing the trajectory picture. The internal noise level during this portion of the flight will be low, governed only by the noise level of the on-board equipment. This takes the reverberation characteristics of the space cabin into consideration. For example, it might be assumed that the ambient noise level of the on-board air conditioning and other sound-producing systems is equal in magnitude to that of the Mercury space capsule air conditioning system. This is 87 db. For comparison, it would correspond to the sound level of a loud conversation.

The effect of a micrometeorite hail should not be ruled out here, but this phenomen is expected to be only transient in nature. Instantaneous response of the structure might be quite high. However, the long-time average structural response would be negligible.

STATIC FIRING AND LAUNCHING

Since no experimental data will be available for some time on the proposed Saturn booster system, it is necessary to make some estimate of the expected noise level external and adjacent to the manned space cabin.

It is known that the rms sound pressure level for the Atlas missile booster system providing 470,000 pounds of thrust is about 141 db. This is at a point 75 feet above the rocket engine exhaust, and would correspond to the location of the missile's nose cone. Allowing 3 db for atmospheric conditions and other uncertainties, this external value would be 144 db.

For a cluster of rockets, the total rms sound pressure level (SPL) of the Saturn system is to the Atlas system SPL, approximately, as the square root of the ratio of their thrusts. Therefore,

$$p_t^2 = \frac{(1.5 \times 10^6)}{0.47 \times 10^6} \; p^2 \text{ ATLAS}$$

$$p_t^2 = 3.2 \; p^2 \text{ ATLAS}$$

Therefore, at 75 feet,

$$SPL_{(SATURN)} = 20 \log_{10}\left(\frac{3.2 \, p^2 \text{ ATLAS}}{p^2_{Ref}} \right)$$

$$= SPL_{ATLAS} + 20 \log 3.2$$

$$= SPL_{ATLAS} + 10.1 \quad \underline{db}$$

However, the space cabin is located 175 feet from the exhaust plane. Therefore, the level at 175 feet would be

$$SPL_{175'} = SPL_{75'} - 20 \log_{10}\left(\frac{175'}{75} \right)$$
(SATURN)

Substituting from above:

$$SPL_{175'} = (SPL_{ATLAS} + 10.1)_{75'} - 7.4$$

$$= (144 + 10.1) - 7.4 = 146.7 \text{ db}$$

or not much higher than that of the Atlas system. This agrees rather well with the empirical predictions of Reference 1. These are based upon engines up to 200,000 pounds thrust extrapolated to those of higher propulsive power and contain data which applies directly to Project Mercury boosters. Although this estimate is subject to error, it is believed that it represents a fairly reliable figure.

Based upon transmission loss data through the thin-walled Discoverer capsule and through other nose-cones of G-E construction, a safe estimate for noise attenuation through the APOLLO capsule may be considered to be about 20 db. This is a conservative figure for over-all attenuation, considering constant octave level band input. Using this number as a first approximation for the transmission loss of sound travelling into the interior of the space cabin, cabin noise levels should not rise above 127 db during the launch phase of the flight. The spectral distribution externally will probably be similar to that of the Mercury space capsule or that of the Big Joe capsule, Figures XI-9-1 and XI-9-2. Most probably the spectrum will approximate that of figure XI-9-1, since larger boosters would furnish more power in the lower frequencies. Note that in these figures the over-all noise reduction is 32 db and 40 db respectively. The reduction at the lower frequencies is based on the unique conical construction of the Mercury system, furnishing a high degree of panel edge restraint and over-all panel stiffness. At the higher frequencies, the mass effect of this construction is effective in reducing noise transmission. Adequate damping of the panels is probably achieved by the use of damping tape or urethane foam insulation.

Consideration of the apparent multiple thin panels of the APOLLO vehicle ranging in thickness between 0.032 and 0.040 inches of aluminum, the transmission loss, T.L., may be estimated from the empirical "mass law":

T.L. = 23 + 14.5 $\log_{10}$ m db where m is the "mass" of the partition in pounds per square foot.

Reference 1. Hilton, D. A.; Mayes, W.A.; and Hubbard, A. H.; Noise Considerations for Manned R/V's etc.

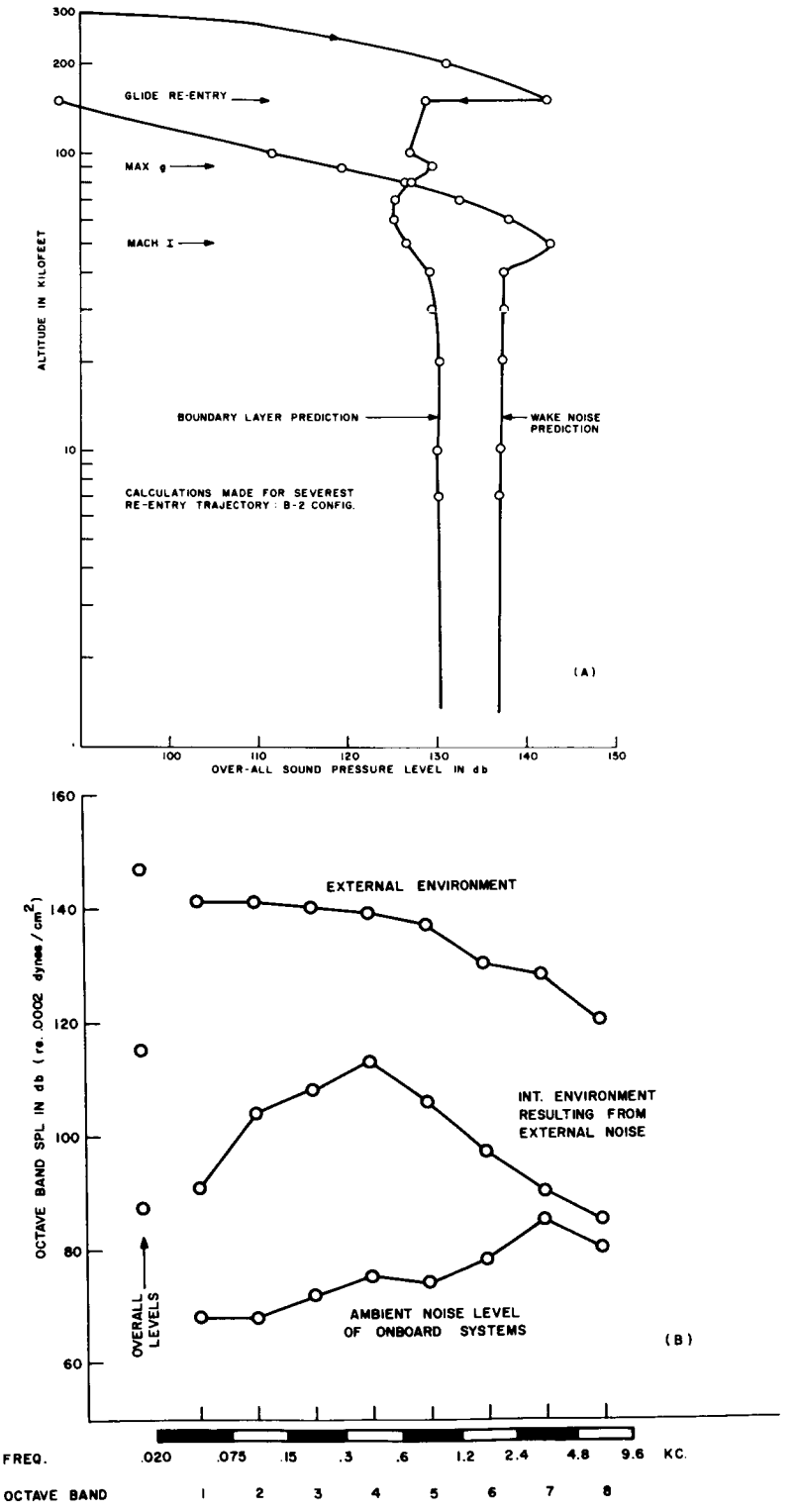


Figure XI-9-1. Measured external and internal environment, Mercury space capsule

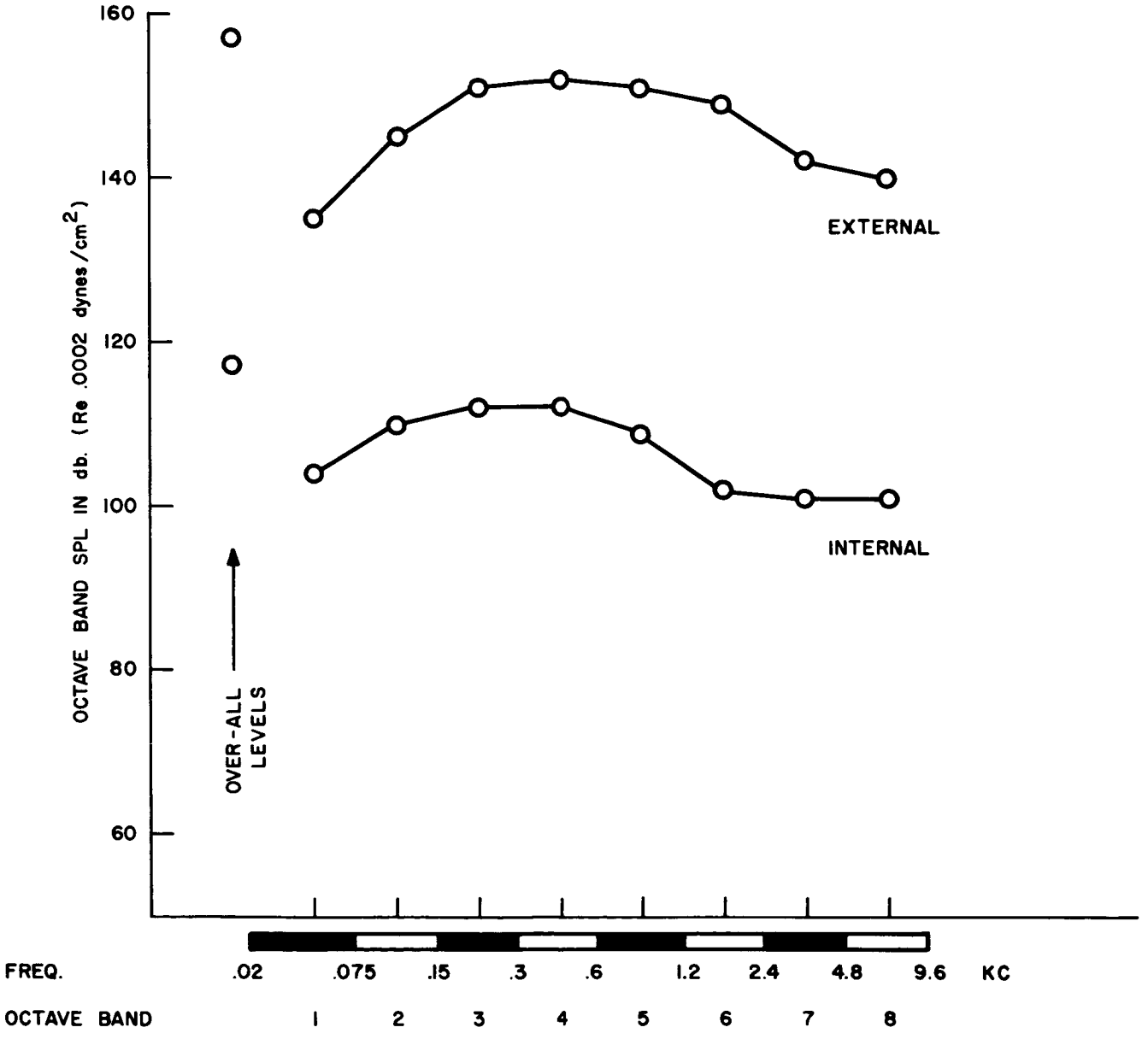


Figure XI-9-2. Measured external and internal environment, Big Joe capsule

Therefore for a one square foot panel of 040 aluminum:

$$m = \frac{.040"}{12} \times 1 \text{ sq ft} \times 0.100 \times 1728 = 0.58 \text{ lbs/sq in}$$

$$T.L = 23 - 3.5 = 19.5 \text{ db.}$$

The behavior of a single wall is shown ideally in Figure XI-9-3. A double wall behaves as in Figure XI-9-4. At a particular frequency (fd) transmission loss drops because the air space, "d", between the walls acts as a spring whose stiffness varies inversely as the panel spacing. The panels behave as two masses, one on each end of the spring. The value of this loss depends upon damping in the cavity or in the panels. In this configuration, since the panels are neither parallel nor of the same thickness, the behavior can be quite complex. For multiple construction, i.e. three or more panels in series, the complexities are even greater. For this reason, the engineering design should be based on measured values of transmission loss. From the above discussion, it is evident that the transmission losses through the thin panels during launch are at least 20 db.

Noise reduction through the ablation shield, based on the Mark II configuration (heat sink) and through the RVX and Mark III configurations, are higher than 20 db. Measured values run from 23 to 28 db over-all (flat spectrum input).

From viewing many oscillograph records of missile flights, it is seen that the internal noise levels during launch last for a very short time, at most five seconds. The significant noise regimes during powered flight are those of Mach I and maximum dynamic pressure (or maximum q). From the predicted dynamic pressure data during the boost phase of this flight, the boundary layer noise external to the manned cabin has been plotted as a function of the dynamic pressure in Figure XI-9-5. The estimated noise is based upon sub-sonic tests by Willmarth in which the rms pressure fluctuations are equal to 0.006g. Then,

$$SPL = 20 \log_{10} \frac{(0.006\ g)}{P_{ref}}$$

Where $P_{ref} = 4.177 \times 10^{-7}$ psf.

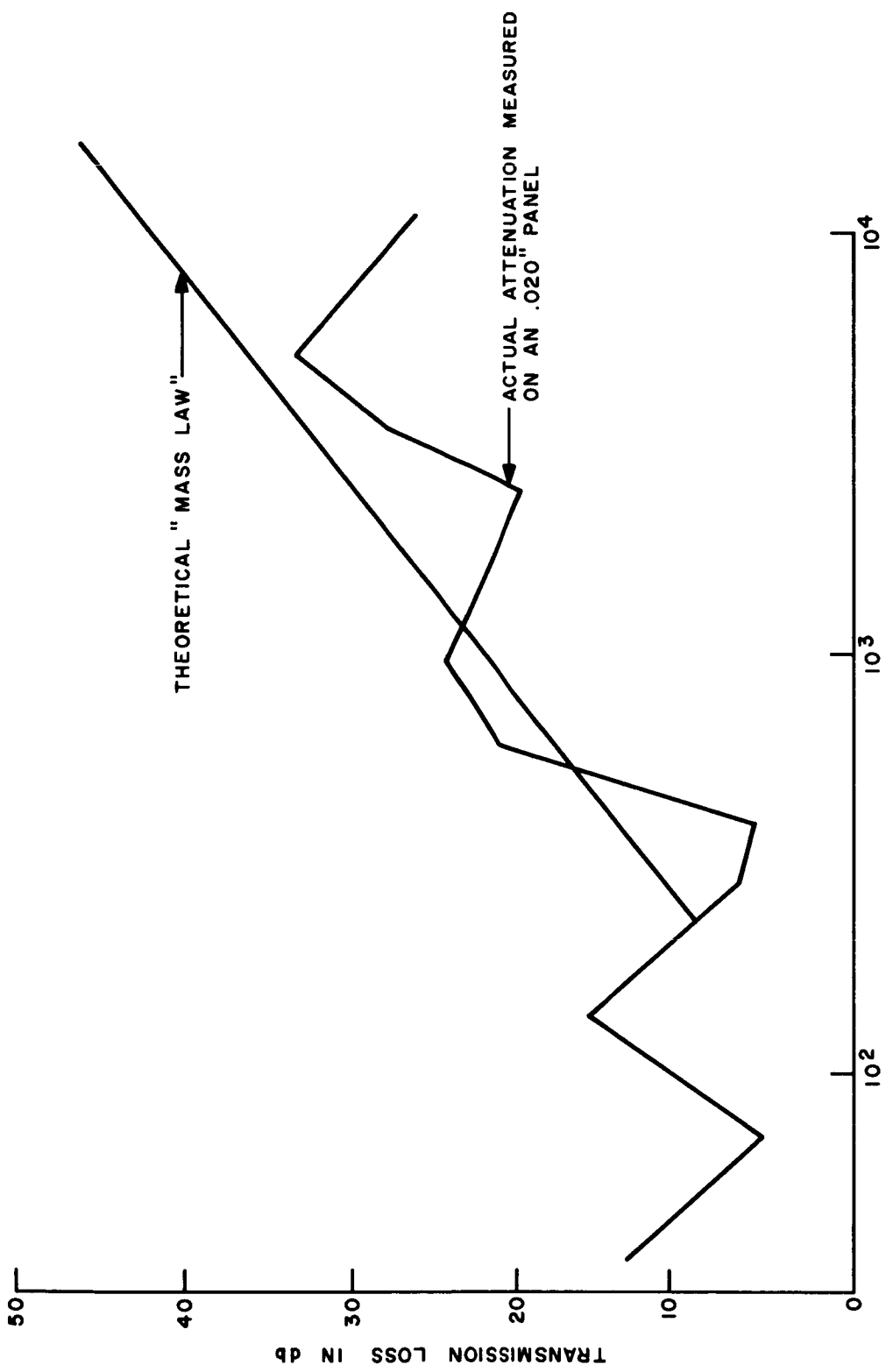


Figure XI-9-3. Transmission loss versus frequency, single homogeneous panel

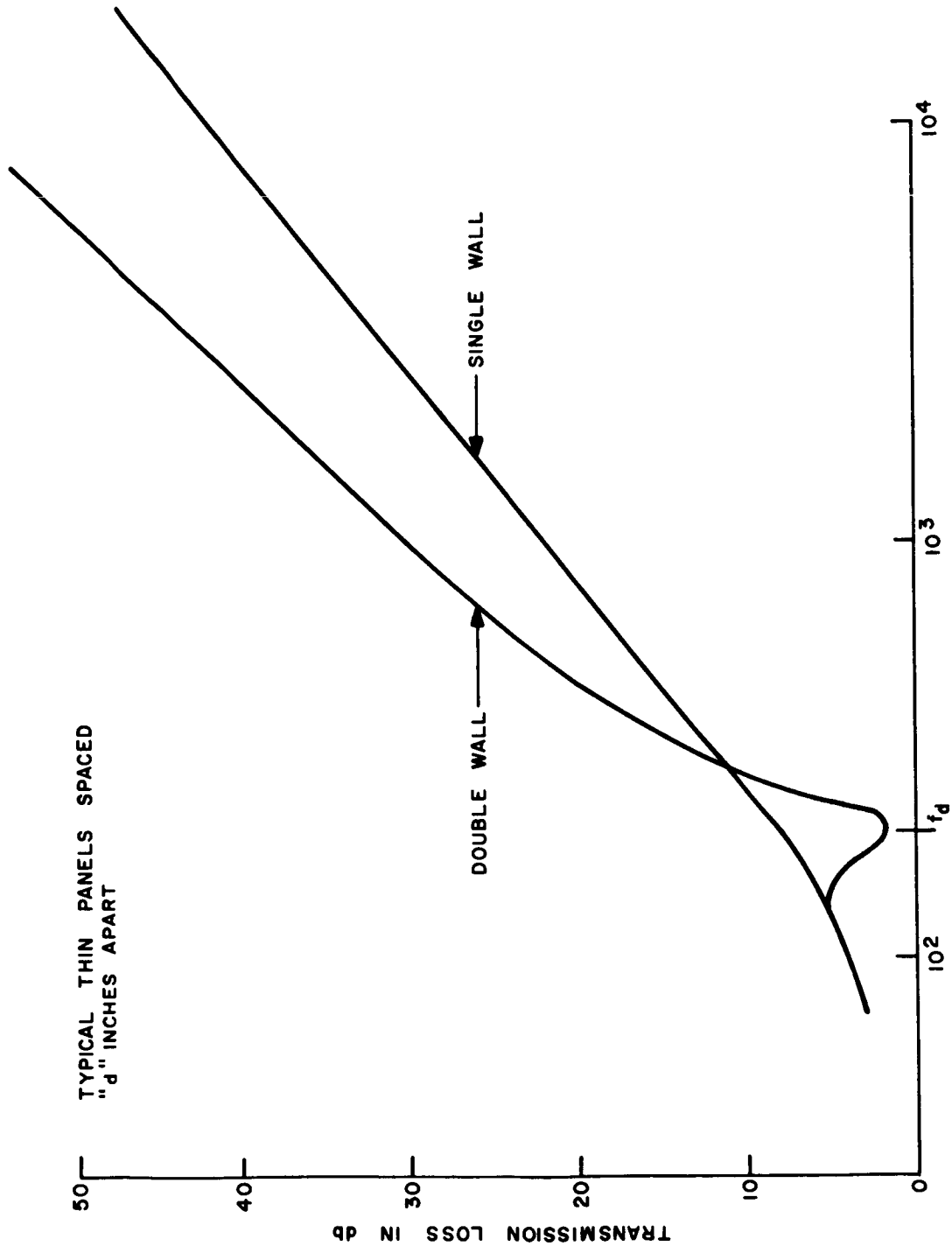


Figure XI-9-4. Transmission loss versus frequency, double-wall versus single-wall construction.

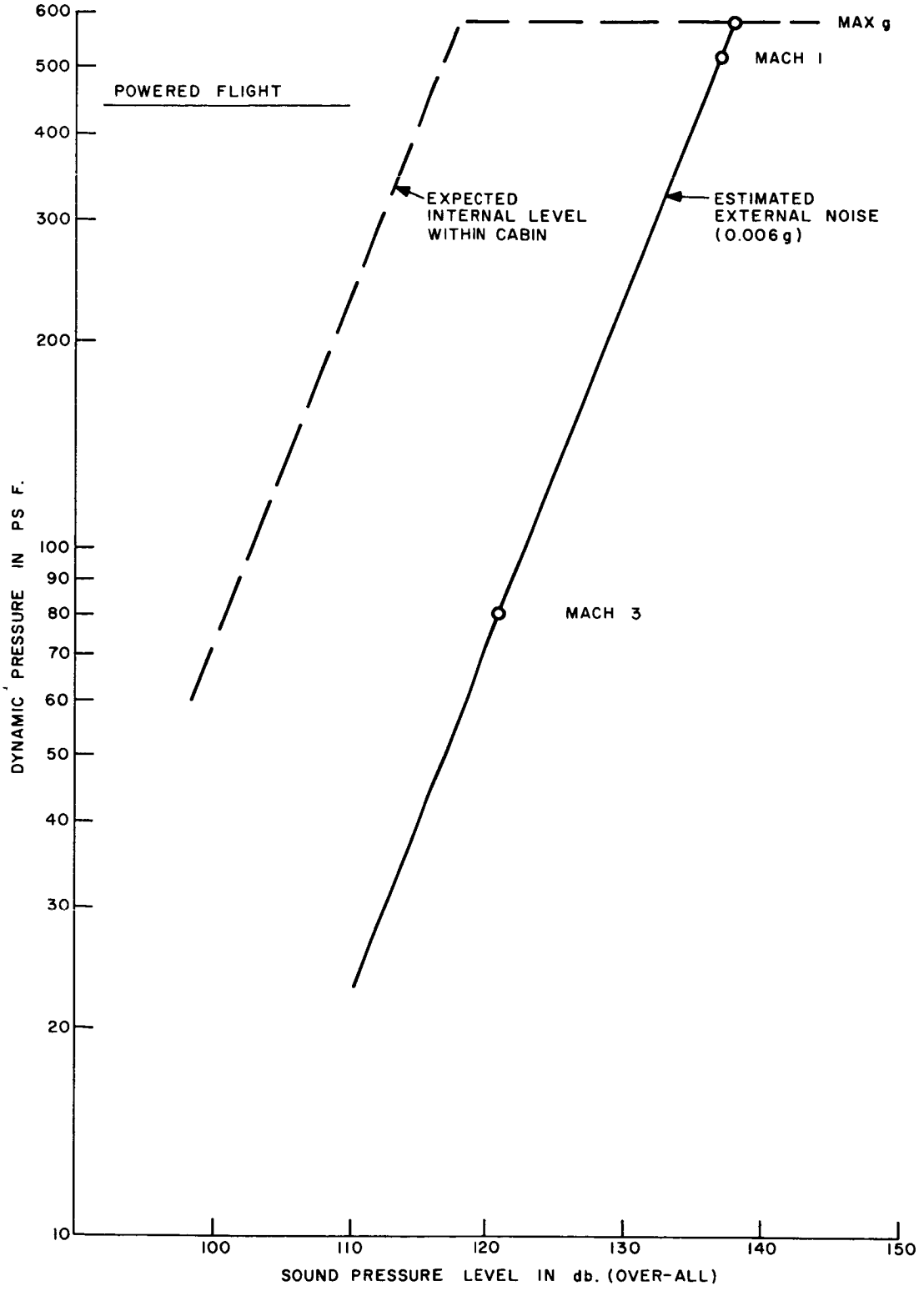


Figure XI-9-5. Estimated external noise, aerodynamic boundary layer turbulence

Application of this equation to Mach number ranges of nearly 6.0 produce good results and so this equation will be retained.

Figure XI-9-6 shows the variation of boundary layer noise with Mach number. As presented, this is an external estimate and it might be assumed that the internal cabin values might be 20 db less than this curve. This is not the case. As the speed of the missile increases, the noise spectrum shifts toward the higher frequencies, peaking at the high Mach numbers. However, as shown in Figures XI-9-3 and XI-9-4, there is more attenuation or noise reduction at the higher frequencies, so that the internal noise is further reduced. Figure XI-9-7 shows the internal measured spectra of the Big Joe vehicle during powered flight at two different Mach values having the same dynamic pressure. This produces, theoretically, the same external over-all noise level but, internally, a reduction of 3.3 db or a reduction in intensity of more than half. It should be noted that these are internal spectra; the external spectra would also be expected to shift in this manner and probably by a greater amount.

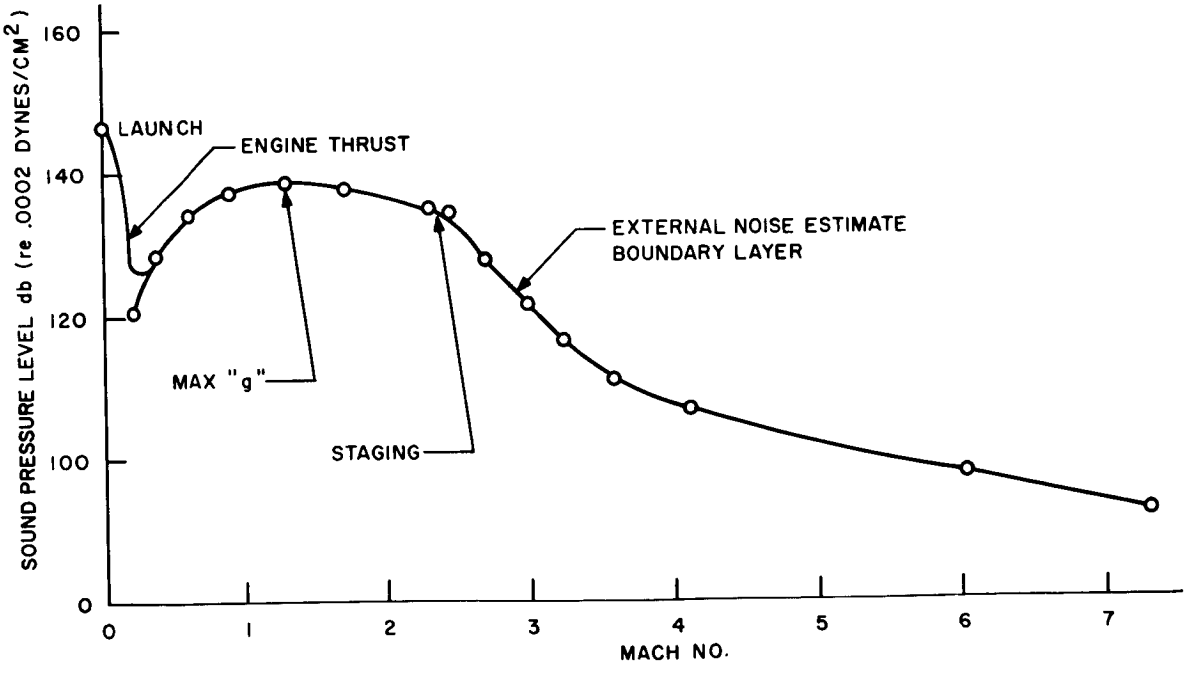


Figure XI-9-6. Effect of Mach number upon aerodynamic boundary layer noise during powered flight, APOLLO vehicle

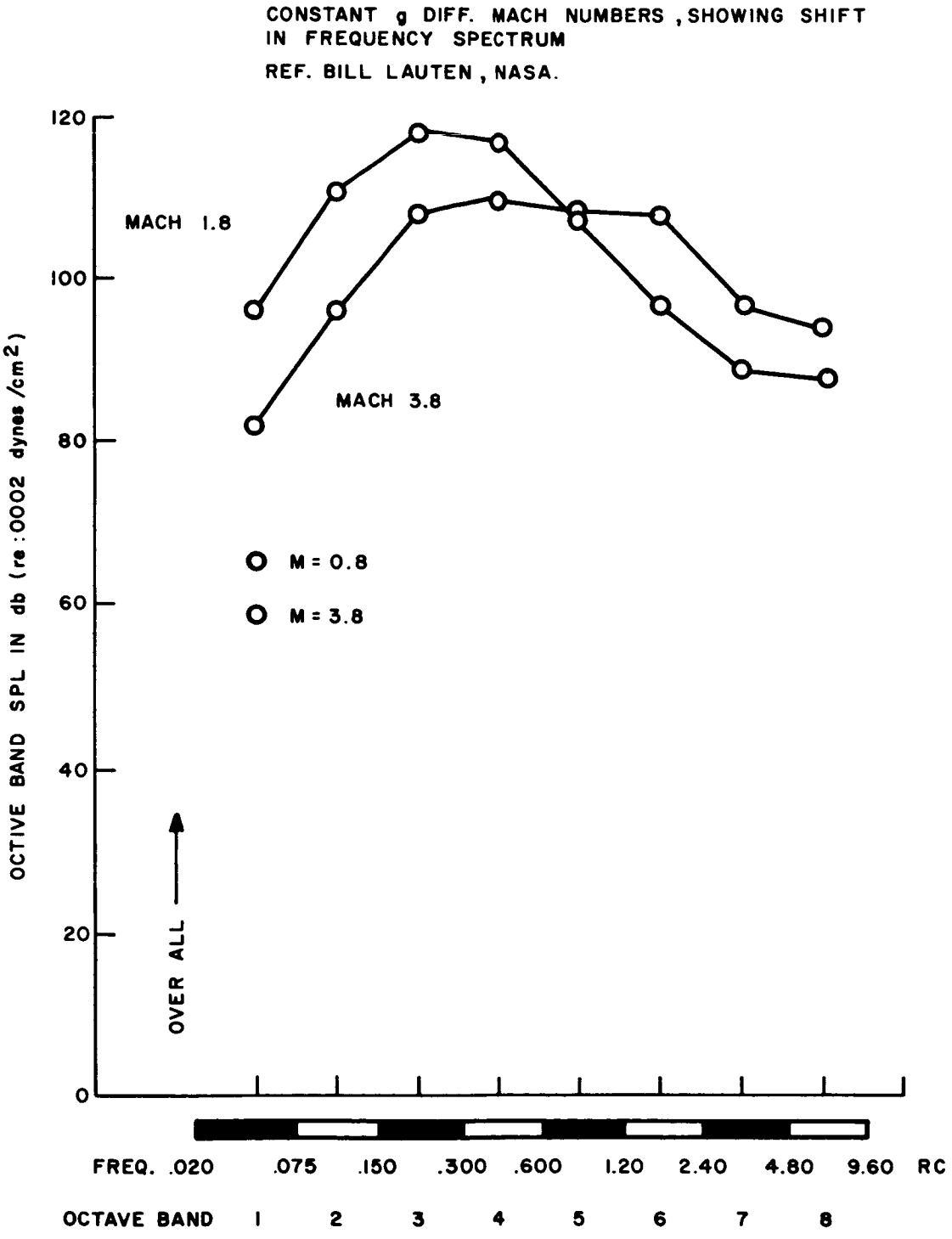


Figure XI-9-7. Measured internal data, Big Joe capsule powered flight

RE-ENTRY

The severest re-entry trajectory was chosen for the B-2 configuration in order to arrive at that combination of velocity, pressure, density, and other parameters which would produce the maximum sound pressure. This information was fed into a digital computer which corrected for, among other factors, Mach number compressibility effects and local flaw conditions. As part of the necessary input information for the turbulent-wake noise predictions, a plot of the base-pressure to free-stream pressure ratio versus Mach number was made available by the Aerodynamics Operation. Furnished with the necessary input data and the appropriate program, the computer printed the necessary data points along the re-entry trajectory.

Figure XI-9-8 shows the external noise prediction for boundary layer and wake excitation in decibels as a function of the re-entry altitude. Cabin levels are assumed to be 20 db below these figures.

CONCLUSIONS

The main sources of noise for a rocket-powered re-entry vehicle are the engines during static firing and launch, the aerodynamic boundary layer during the high-dynamic-pressure portions of the flight and, finally, the turbulent atmosphere during re-entry.

Noise predictions for all three phases of the flight have been made. Data has been based, in part, by the Mercury vehicle data correlations. Other information, particularly the re-entry phase, have been based upon information gained from a whole series of RVX, Mark II and Mark III flights.

A value of noise attenuation through structure is taken to be 20 db. This is admittedly a conservative figure which should suffice until the structure has been more closely defined in detail.

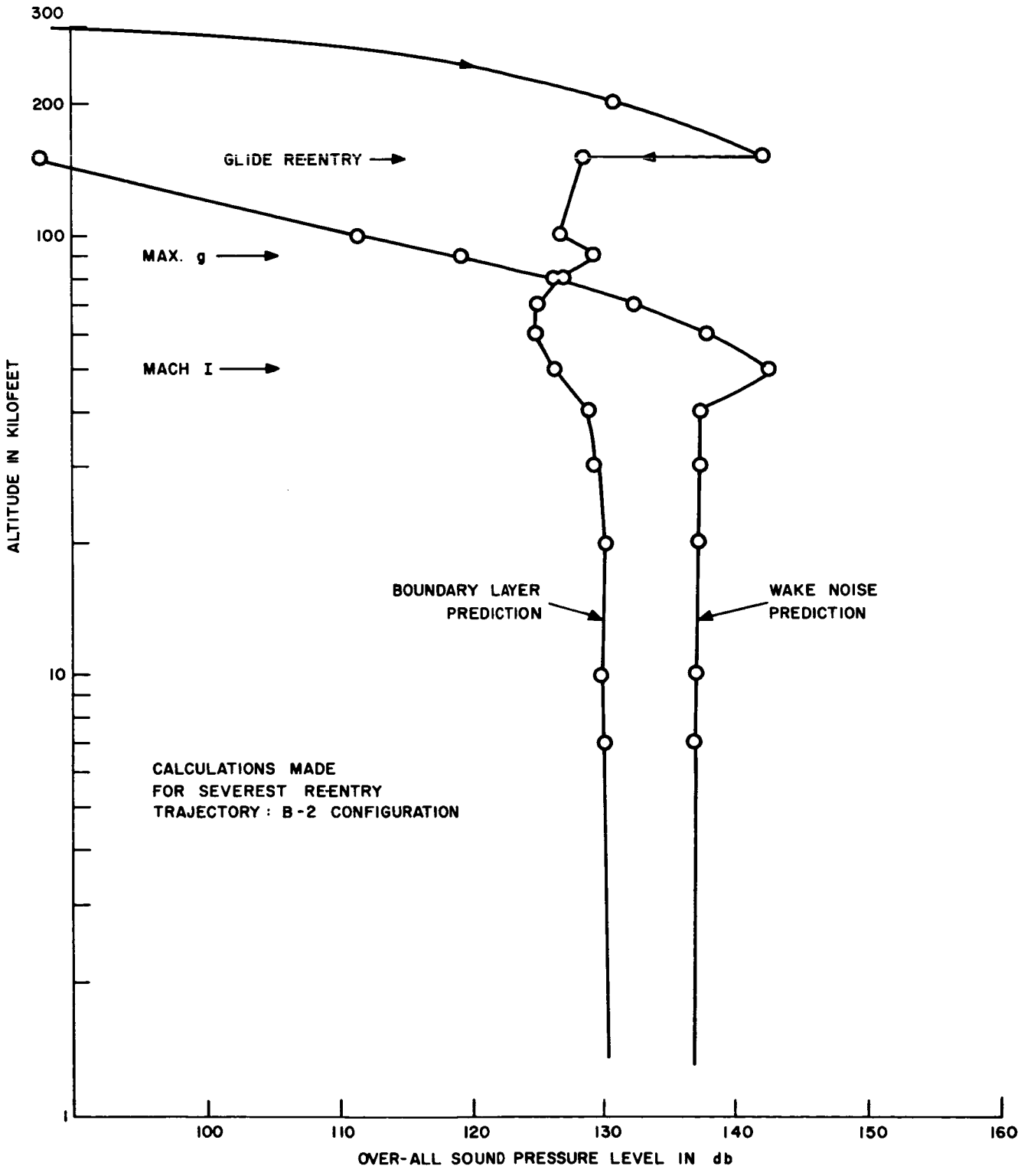


Figure XI-9-8. External noise prediction for boundary layer and wake excitation, APOLLO vehicle

10. ATMOSPHERE SELECTION

INTRODUCTION

The objectives of the environmental control study are (a) to select the vehicle internal atmosphere and (b) to provide a preliminary design of the system for maintaining it. The system must be reliable and, ideally, should be light in weight and require little volume and power.

SELECTION OF THE ATMOSPHERE

Ground Rules for Atmosphere Selection

The "ground rules" used in selecting the internal atmosphere of the vehicle are:

(a) The atmosphere must, in accordance with the "shirt-sleeve environment" philosophy, produce minimum physiological stress to the crew.

(b) In the event the vehicle is punctured, it should provide a viable atmosphere for as long as possible and, in the event the pressure vessel integrity of the vehicle is lost and it becomes necessary to maintain the crew at a reduced pressure by means of a secondary pressurization system, then the total pressure change should not cause aeroembolism.

(c) The atmosphere should not, insofar as possible, increase fire hazard by increasing the rate of combustion.

(d) The atmosphere selected should be one which is capable of being maintained with a minimum weight and volume system, requiring minimum electrical power.

Unfortunately, some of these objectives tend to place contradictory requirements upon various parameters and, therefore, it becomes necessary to trade-off one objective versus another.

Some Considerations of Oxygen Partial Pressure

The most essential ingredient in the atmosphere is oxygen. Gross permissible limits can be placed upon the oxygen partial pressure, the lower limit being the one which just maintains the alveolar pO_2 above the hypoxic level and the upper limit being the one which, if exceeded, will result in oxygen toxicity(1). Within this range, various factors must be traded off. The possibility of atelectasis, especially during acceleration(1), the fact that the leakage weight is a linear function of the partial pressure, and the sharp increase in combustion rate with oxygen partial pressure, all suggest that the pO_2 be kept close to the lower permissible level. Contradicting this is the desire to maintain as high a pO_2 as possible in order to provide the maximum useful time to the astronauts to take whatever action is indicated in the event of rapid leakage due to meteorite puncture or some component or structural failure. Although, qualitatively, the effects of the pO_2 are as indicated above, quantitatively the effects are dependent on whether or not a diluent gas is used and, if so, on which gas is used and upon its partial pressure. Therefore, the quantitative trade-off on oxygen partial pressure will be made after the question of the diluent gas is investigated.

Why a Diluent Gas?

It was decided to use a diluent gas for the following reasons:

1. From the considerations previously discussed on pO_2, it appeared likely that a rather low oxygen partial pressure would be used. For a given cabin heat rejection system and a fixed thermal load, the total fan power was investigated. From Figure XI-10-1, it can be seen that the fan power required is an inverse exponential function of total pressure. Comparing two points on the curve, it can be seen that, for an atmosphere consisting of 180 mm O_2, 9 mm H_2O, and 4 mm CO_2, the fan power required is $87.5/25 = 3.5$ times as high as it would be if the total pressure were raised to 360 mm by adding 180 mm of nitrogen. For the heat loads calculated for APOLLO, this means a reduction from 560 to 160 watts or a difference of 400 watts continuous.

References:

(1) Chapter IV, Biological Requirements

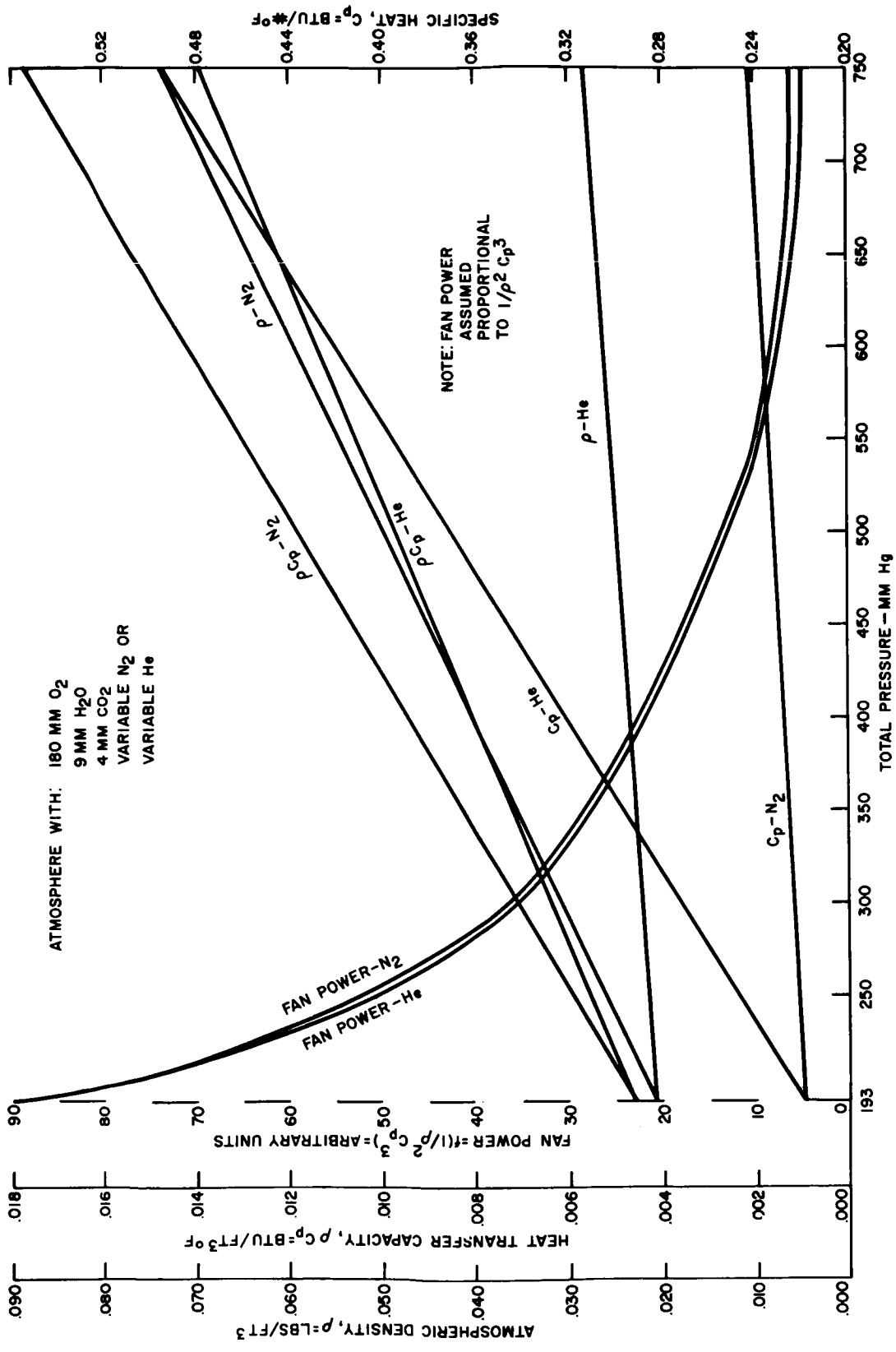


Figure XI-10-1. Atmospheric properties and blower power as a function of atmospheric composition and pressure

2. It has been shown that, for a fixed pO_2, combustion time increases—and therefore fire hazard decreases—as the partial pressure of an inert diluent gas is increased. Figure XI-10-2 indicates the effects of pO_2 and pN_2 on combustion time. The raw data used to prepare this curve were taken from a curve in a report by Simons and Archibald[2] and are redrawn and extrapolated considerably. These data were taken in the Earth's gravitational field; consequently, they are not quantitatively applicable to the zero-g condition where natural convection is absent. Quantitatively, the inhibiting action of the diluent gas upon flame propagation should be considerably enhanced in the zero-g field.

These considerations indicate that the diluent partial pressure should be as high as possible; however, other considerations place a limit upon this value. One consideration is the occurrence of aeroembolism in the event of rapid cabin depressurization; another is that the leakage weight of diluent gas is linear with its partial pressure; a third is that the vehicle structure weight must increase when the internal atmospheric pressure exceeds some (as yet undefined) level.

Aeroembolism occurs when the total volume of gas released in bubble form within the body, due to decreased gas solubility in both the body fat and blood as a result of decreased total pressure, exceeds a critical value. It has been shown that the symptoms of aeroembolism are a function of the total amount of gas released which, in turn, is a function of the pressure change ratio. The severity of damage, however, is not only a function of the pressure change ratio but is a function of the initial partial pressure of the diluent as well[3]. The amount of gas released is, in addition, not only a function of the pressure change ratio but is also a function of the diluent gas used[1]. Consequently, for each diluent gas there is a permissible pressure change ratio. This will be discussed in more detail when diluent gases are compared.

The quantitative weight assignable to leakage is a function of the gas used, as well as partial pressure, so this will be evaluated quantitatively when diluent gases are compared.

References:

(1) Chapter IV, Biological Requirements

(2) Simons, D. G. and E. R. Archibald; "Selection of a Sealed Cabin Atmosphere"

(3) Guyton, A. C., "Textbook of Medical Physiology"

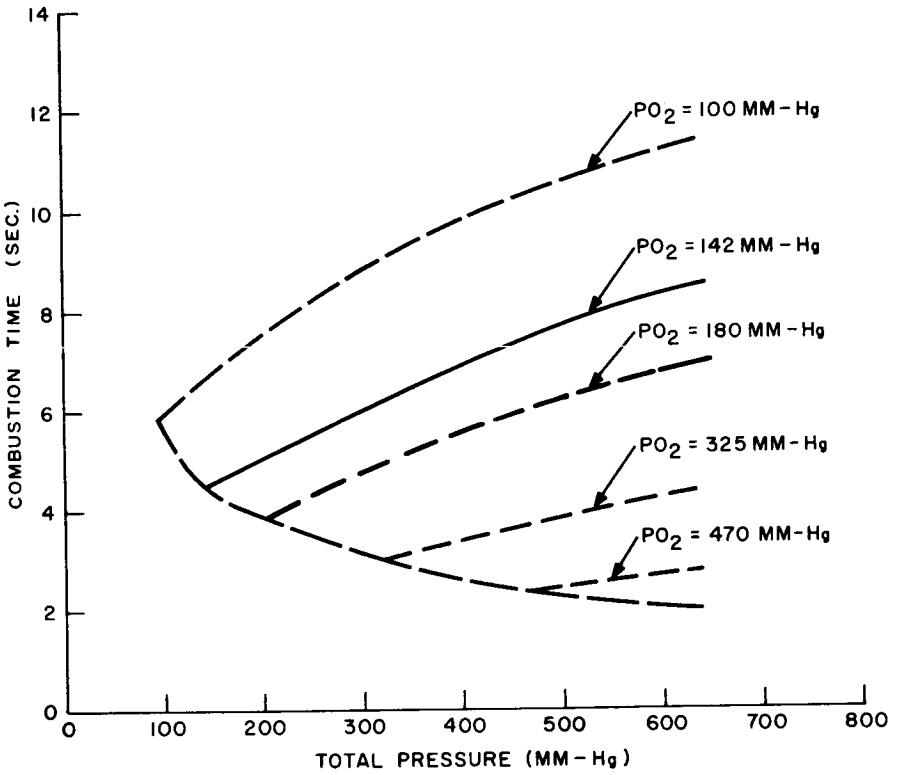


Figure XI-10-2. Flammability of paper strips versus total pressure at different oxygen partial pressures

A disadvantage to using a diluent gas is that it will be necessary to regulate the rate at which oxygen is admitted to the cabin by means of a partial-pressure regulator rather than by means of a total-pressure regulator. This, however, is only a slight disadvantage, as will be shown later when the atmosphere sensing and electrical control subsystem are discussed.

What Diluent Gas Should Be Used?

Of the diluent gases suitable for use, the field can be narrowed rather quickly to nitrogen, helium[1], and neon. Practically no information is available on the long-term physiological effects of neon as a diluent gas. In addition, this gas requires somewhat more fan power than either helium or nitrogen and is only slightly better than nitrogen from the aeroembolism point of view. Consequently, for a detailed

Reference:
(1) Chapter IV, Biological Requirements

comparison, only nitrogen and helium will be considered. The primary advantages of helium are that it requires somewhat less heat-exchanger fan power than nitrogen (see Figure XI-10-1) and (assuming previous denitrogenation of the crew) permits a higher pressure-change ratio than nitrogen before the symptoms of aeroembolism develop. The disadvantages of helium are:

(1) For a given vehicle hole size, the weight of the system used to store the helium leaked is greater than that of nitrogen (assuming cryogenic nitrogen storage). Figure XI-10-3 shows the leakage weight comparison for the largest equivalent hole size expected in the APOLLO vehicle.

(2) In the event of a puncture of the vehicle, it will, for the same initial pO_2 and total pressure, take less time with He than with N_2 as the diluent for the pO_2 to reach the lower critical value. As can be seen in Figure XI-10-4, for an initial 360 mm total pressure, an initial pO_2 of 180 mm, and a one sq. in. hole, it will take 54 seconds for the pO_2 to decrease to 90 mm with N_2 and 42 seconds with H_e. Consequently, with N_2 as a diluent, the crew will have 29 percent more useful time to take whatever corrective action is indicated than they would with He as a diluent.

(3) As can be seen from the data plotted on Figure XI-10-5, which compares leakage weight in lbs/hr versus hole size for helium-oxygen and nitrogen-oxygen mixtures, the leakage loss, through repairable-type holes, will be considerably greater for the helium-oxygen mixtures. The hole sizes contemplated on this plot are such that it may take some time to detect and repair them. By comparing curves 4 and 6 of Figure XI-10-5, it can be seen that, for a hole diameter of 0.1 inch, the loaded weight of nitrogen leaked is 3.5 lbs/hr and for helium 5.0 lbs/hr. This, however, is not the complete story. Because the gas leaked is homogenous and the volumetric leakage weight depends upon sonic velocity (which in turn depends upon the mixed gas properties), the amount of oxygen leaked with helium as the diluent is greater than when nitrogen is the diluent. This difference can be seen to equal (by comparing curves 1, 2, 3 and 5 of Figure XI-10-5) 0.5 lb/hr more O_2 when helium is used than when N_2 is used. This extra O_2 has a loaded weight of 0.9 lb. Consequently, the difference in leakage is 3.5 lbs/hr versus

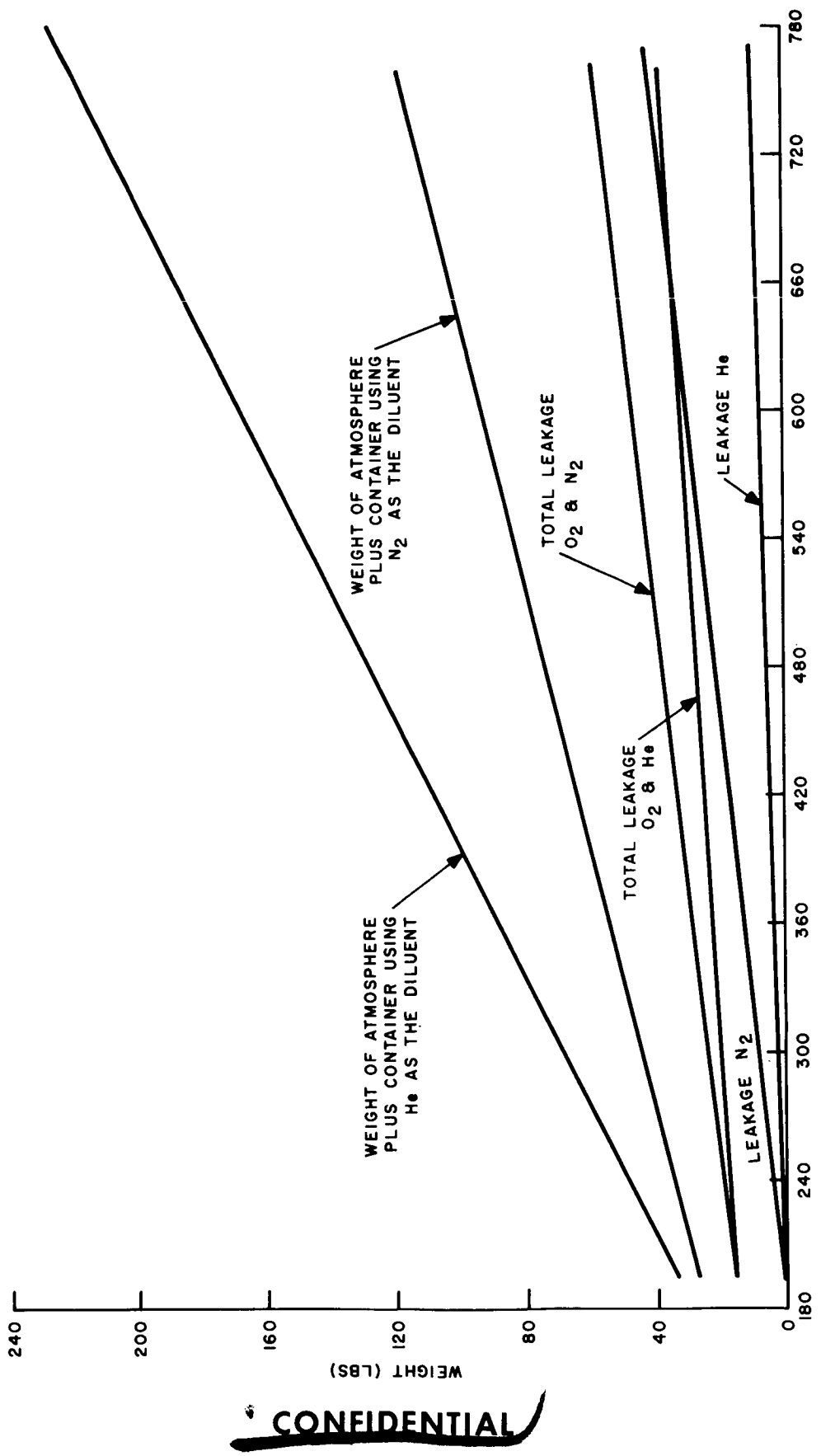


Figure XI-10-3. Leakage weight penalties, nitrogen (N_2) versus helium (He)

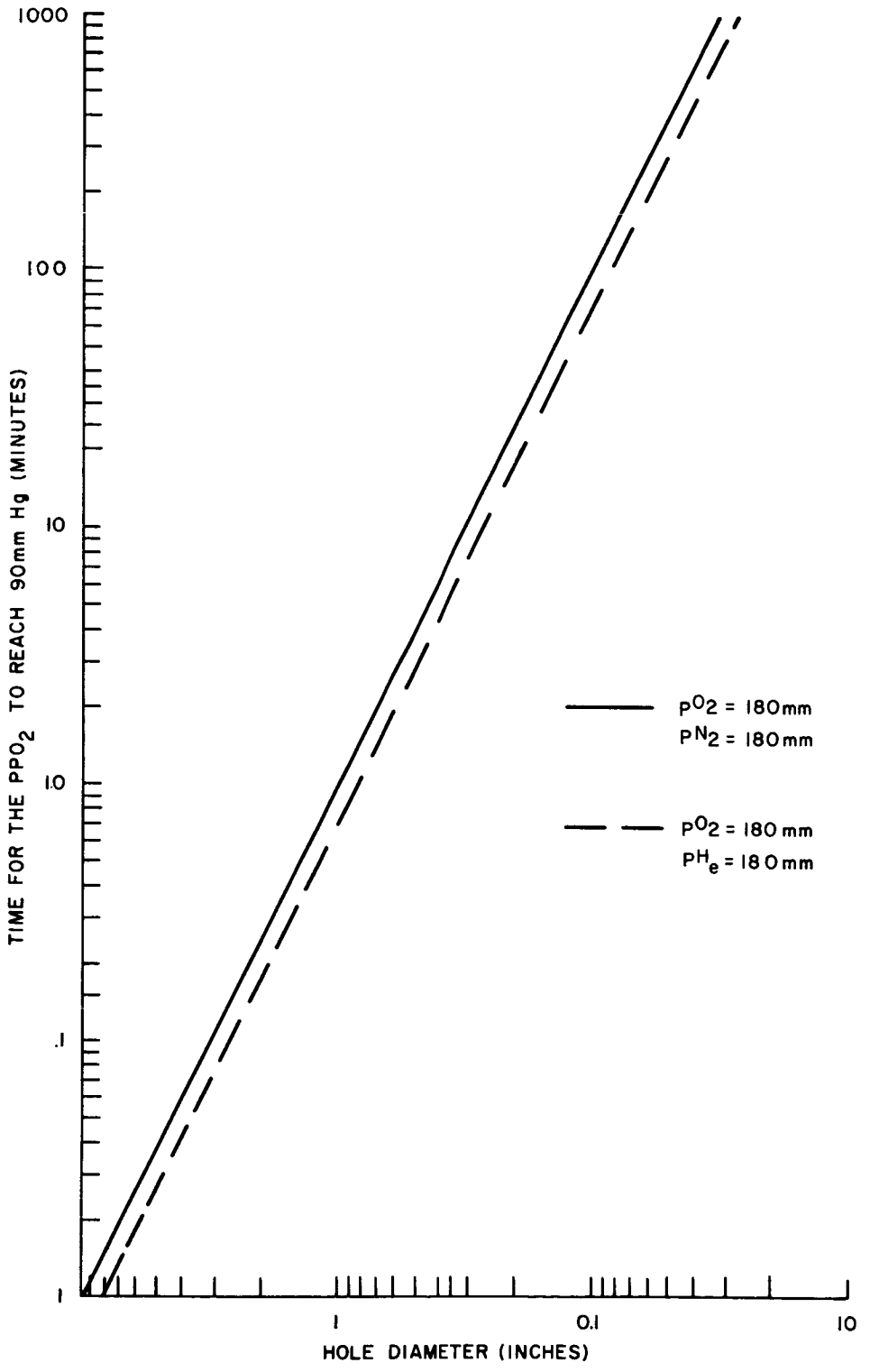


Figure XI-10-4. Orifice area (sq ft) versus time (sec) for oxygen to reach the minimal partial pressure of 90 mm Hg considering both nitrogen and helium as the possible diluent gas at a total pressure of 7 psia and a free cabin volume of 600 cu ft

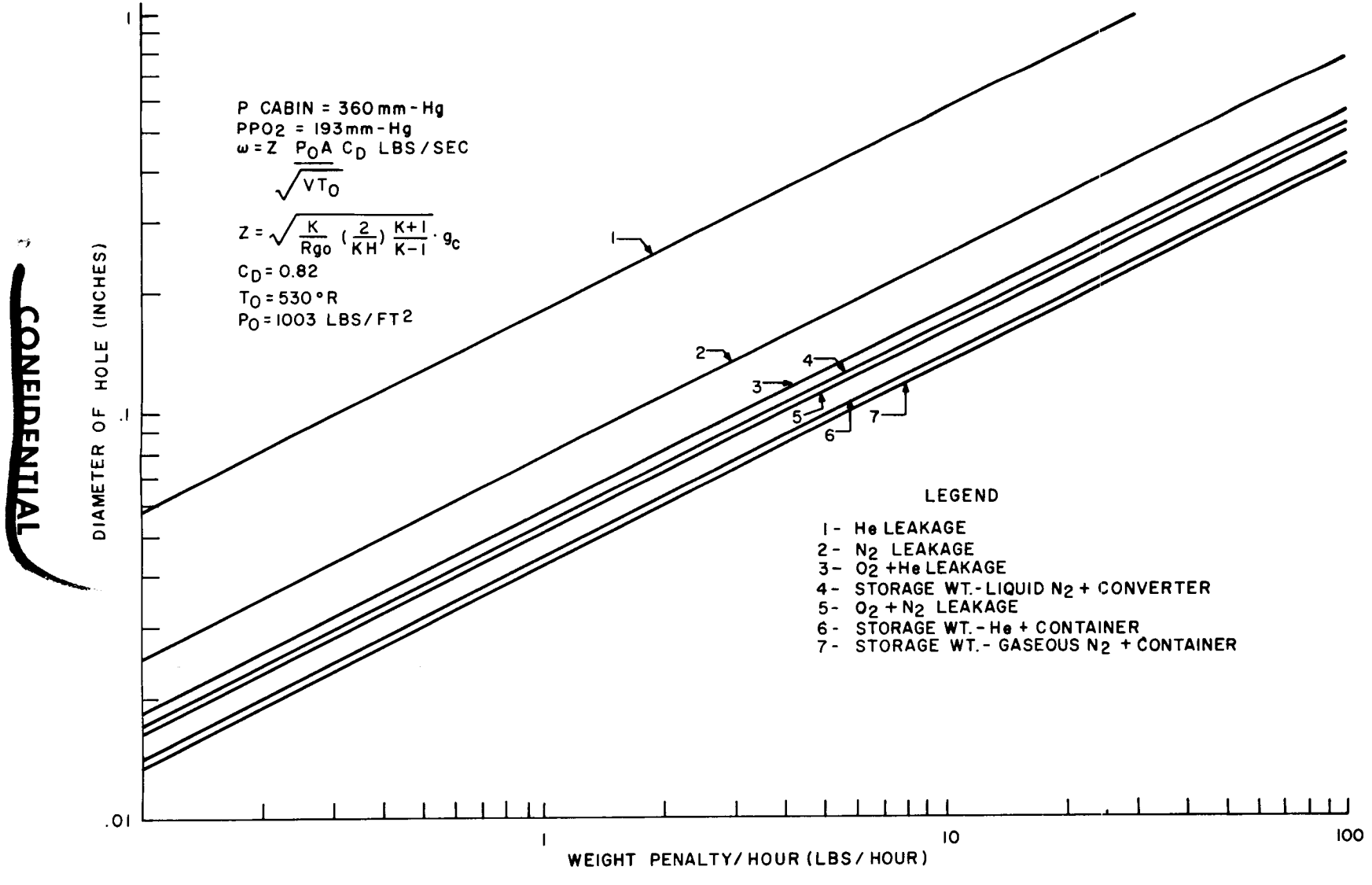


Figure XI-10-5. Leakage weight (lbs/hr) penalties versus hole size for helium-oxygen and nitrogen-oxygen mixtures

5.9 lbs/hr; showing that, for a hole of this size, the loaded weight loss of the He-O_2 mixture is 68 percent greater than it is for the N_2-O_2 mixture.

(4) Some additional, but hard to quantitatively evaluate, disadvantages to the use of helium are its adverse effect upon voice communication and its expected inferiority to nitrogen as a combustion inhibitor in the zero-g environment. In the absence of natural convection and (assuming all blowers are turned off in the event of fire) forced convection, the only way that the O_2 molecules can reach the combustion zone is by diffusion through the gas cap surrounding the conflagration. Because of its lower molecular weight, helium will permit the O_2 molecules to diffuse more rapidly than nitrogen. Tending to offset this disadvantage of helium are its high specific heat and thermal conductivity, which will tend to cool the conflagration and, thereby (qualitatively at least) tend to decrease the combustion rate. However, the increased diffusion rate of oxygen in the helium is considered to be the more significant effect and, as stated previously, the net effect should be an increase in combustion rate with helium as the diluent.

Figure XI-10-6 compares the system weight of He-O_2 and N_2-O_2 mixtures as a function of the diluent partial pressure. Several facts are evident:

1. Total system weight for both diluent gases at first decreases and then increases with increasing partial pressure of the diluent, showing that the total system weight can be decreased by adding a diluent gas to the basic oxygen pressure requirement.
2. For each partial pressure of the diluent gas, the N_2-O_2 system is lighter than the He-O_2 system.
3. For a high total pressure atmosphere, where the advantage of using helium (to prevent aeroembolism in the event of rapid depressurization) is realized, the weight penalty due to leakage is enormous.

If it is assumed that the secondary pressurization system has a total pressure of 187 mm (the lowest pressure that can be tolerated for an O_2 atmosphere and still maintain a normal alveolar pO_2) the safe cabin total pressure to avoid aeroembolism with N_2

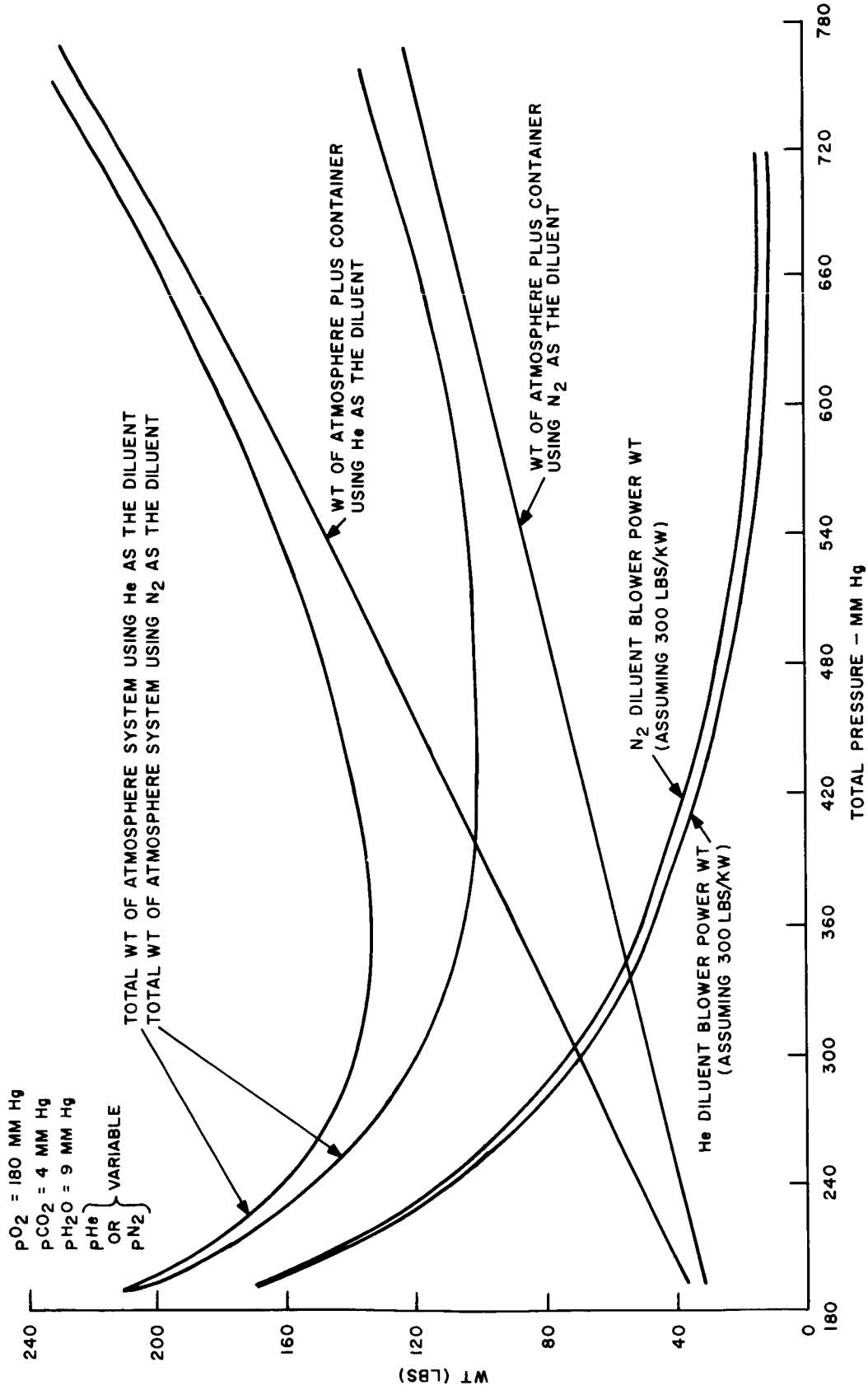


Figure XI-10-6. Weight of make-up gas supply plus blower power versus total pressure; H_e versus N_2

as a diluent, is in the range of 360 mm to 420 mm. If a cabin pressure of about 360 mm is assumed, there will be no advantage to using He as a diluent gas from the point of view of aeroembolism. (This, of course, assumes that the pN_2 of the blood and fat has been equilibrated with the cabin atmosphere pN_2.)

Weighing all these factors, it is obvious that N_2 is by far the better diluent gas.

Summing Up

Summing up, MSVD selects a pO_2 between 170 and 190 mm and a total pressure between 350 and 370 mm. This atmosphere optimally satisfies the basic "ground rules" stated at the beginning of this discussion. In addition, a maximum pCO_2 of 8 mm and a pH_2O of 5 to 15 mm are specified(1).

Reference:

(1) Chapter IV, Biological Requirements

II. ENVIRONMENTAL CONTROL SYSTEM CONCEPTS

INTRODUCTION

The environmental control system is required to maintain the effective temperature, the gaseous constituents, the ion balance, and the particulate matter concentration of the internal cabin atmosphere within predefined limits. Subsystems related to the environmental control system which are included in the following description are the fire-detection and control system and the electrical and electronic cooling system.

EFFECTIVE TEMPERATURE CONTROL SYSTEM

The effective temperature relates the thermal physiological effect of the atmospheric dry-bulb temperature, the atmospheric velocity, density, relative humidity, and surrounding wall temperature. This effective temperature will be maintained at 70 ± 5 F, see Figure XI-11-1. The system selected for rejecting the cabin heat is described with the help of the attached schematic, Figure XI-11-2. The compartment atmosphere is passed through the compartment heat exchanger at a fixed rate by the two blowers shown, either one of which is capable of providing sufficient flow to maintain the cabin at the upper effective temperature limit of 75 F. The atmosphere, in passing through the heat exchanger, is cooled to about 45 F and dehumidified to a dew point of about 42 F. The heat is rejected within the heat exchanger to a liquid coolant which is pumped through the heat exchanger, then through the chassis of the electronic equipment where it absorbs the major portion of the electrical cabin internal heat input. The coolant is then pumped through a radiator, integral with the external skin of the vehicle, where the heat absorbed in the cabin is rejected to space. The bypass valve shown in the coolant circuit is controlled by the compartment thermostat. It directs the flow of coolant either through the compartment heat exchanger or through the bypass, thereby maintaining the cabin effective temperature at the set level. There will always be a flow of coolant through a part of the compartment heat exchanger, as shown, by a

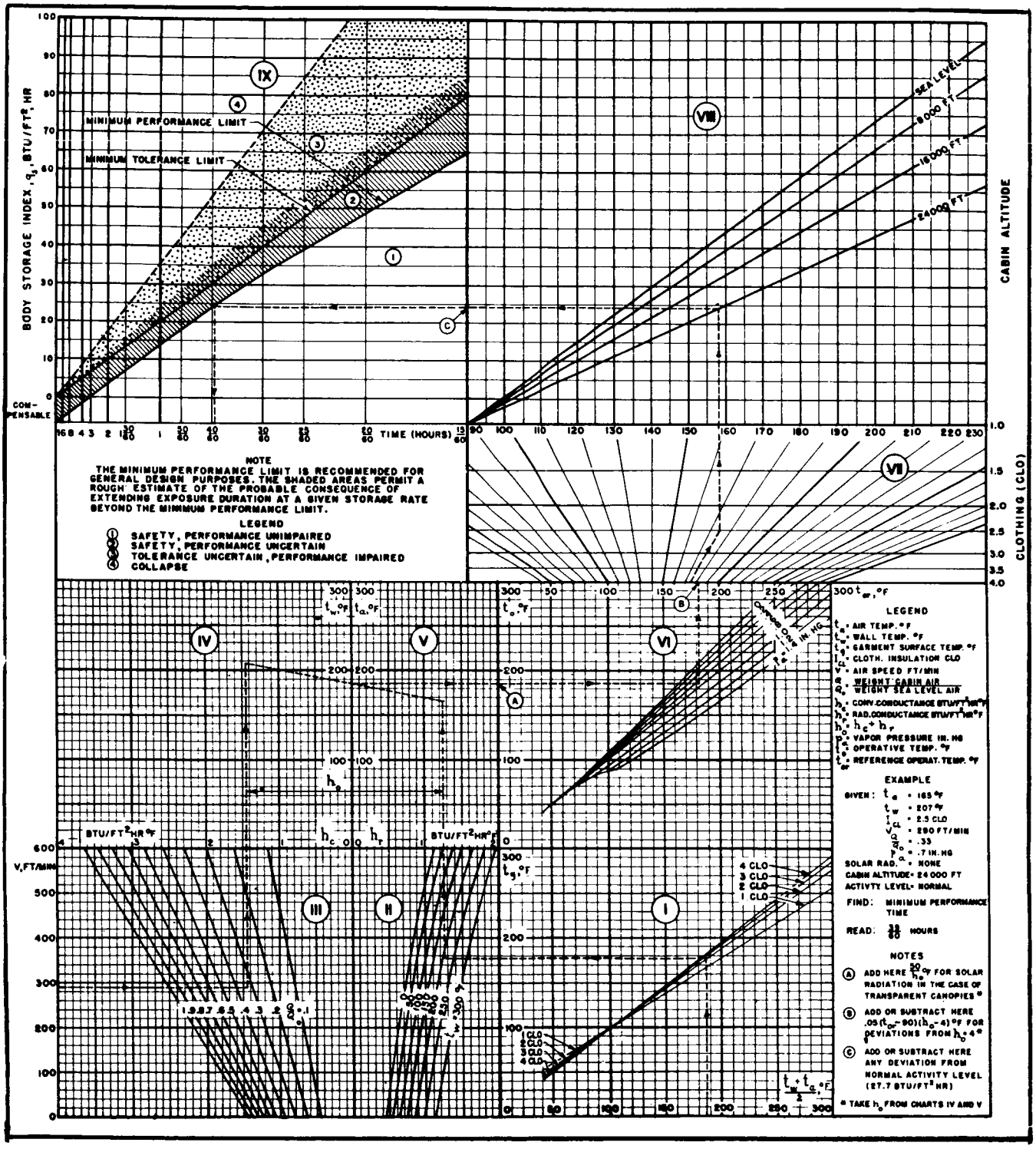


Figure XI-11-1. Hot zone requirements for crew positions in aircraft (ARDCM 80-1)

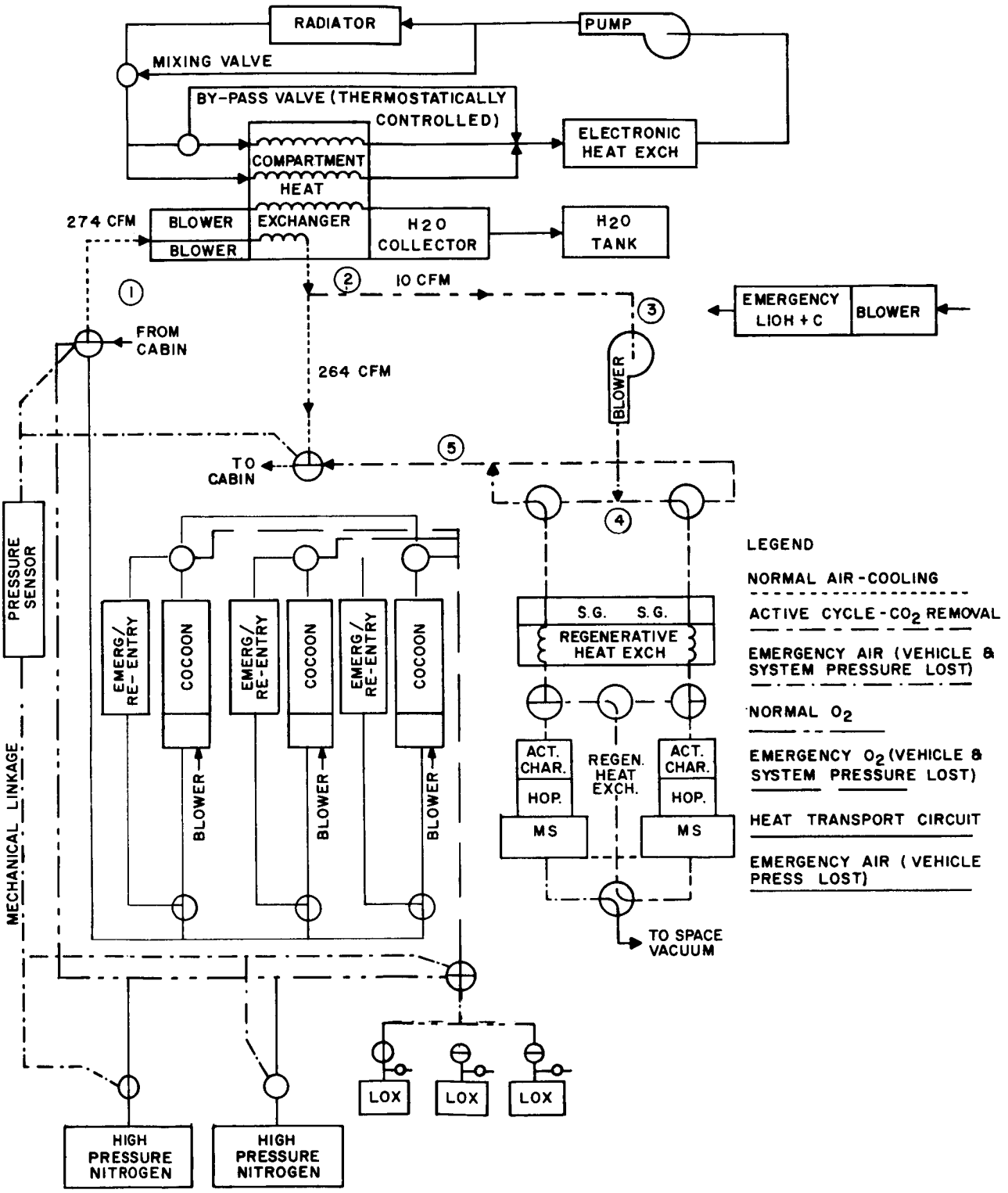


Figure XI-11-2. APOLLO atmospheric control system

parallel circuit. This assures humidity control. The mixing valve shown short-circuits just enough fluid past the radiator so as to maintain the coolant temperature leaving the mixing valve at 30 to 35 F.

The water vapor condensed on the surface of the compartment heat exchanger is collected on a sponge which is squeezed, and the water collected is forced into a storage tank where it is available for whatever use is desired.

In the normal mode of operation, the major portion of the atmosphere leaving the heat exchanger is returned to the cabin. Should the command module depressurize, the vent to the cabin is closed and the air is directed to the secondary pressurization system, shown in the schematic as a cocoon complex.

GASEOUS CONTROL SYSTEM

The gaseous control system has the following primary functions:

(a) To store the oxygen required to replenish that consumed by metabolism and lost due to leakage — and possibly purging — and, in the normal mode of operation, to meter it into the cabin at a rate equal to the consumption rate, thereby maintaining the oxygen partial pressure within predetermined limits. Should the cabin irreparably lose its capability of maintaining pressure without undue leakage, then the oxygen will be supplied directly to the secondary pressure protection system.

(b) To store the diluent gas required to replace that lost by leakage — and possibly purging — and to meter it to the cabin at a rate just sufficient to maintain the cabin total pressure within predefined limits.

(c) To remove water vapor, carbon dioxide, and miscellaneous noxious and toxic gases at a rate adequate to maintain their partial pressures within predefined limits.

The oxygen is stored in the liquid form in four converters, each holding 28 pounds of oxygen. Three converters contain enough oxygen to provide for the maximum expected leakage and the average oxygen consumption of three 80th-percentile astronauts for 14 days. The fourth converter contains a reserve supply for use in the event that one of the other converters fails to operate; the crew consumes more than expected; a purge of the cabin is required; or the leakage rate is greater than expected.

The diluent gas, nitrogen, will probably be stored as a high pressure gas in two containers. Each container holds approximately 11 pounds. The nitrogen will be metered by a total pressure regulator to maintain the cabin pressure at 360 ± 10 mm Hg.

The carbon dioxide will be removed by adsorbing it from the cabin atmosphere on molecular sieves, which are periodically reactivated by exposure to space vacuum. Because molecular sieves preferentially adsorb water to the exclusion of CO_2, the atmosphere to be passed over the molecular sieve bed must first be dried. This is accomplished by passing it through a silica gel bed. The operation of the system is further explained with reference to the schematic. After leaving the valve at point (4), the portion of the atmosphere bled off from that leaving the heat exchanger (having a temperature of 45 F, a dew point of approximately 42 F and pCO_2 of less than 8 mm Hg) is passed through a silica gel bed physically contained within a specially designed air-to-air type heat exchanger. When making the first pass through the silica gel, the air is dehumidified; when making the second pass, it is rehumidified. When passing through the activated charcoal and hopcalite, noxious and toxic gases are removed by adsorption and catalytic combustion, respectively. When passing through the molecular sieve, the CO_2 is removed by adsorption. Meanwhile, the other chemical bed is being reactivated by exposure to space vacuum, as shown. When the core of each valve shown between positions (4) and (5) is rotated 90 degrees counterclockwise from the position shown, the operation of the system is reversed.

The silica gel and molecular sieve are contained within an air-to-air type heat exchanger so that the heat released during adsorption (approximately 1400 Btu per pound of water adsorbed and 400 Btu/lb of CO_2 adsorbed) can readily be transferred to the bed where desorption is occurring. The LioH cannister shown between positions (4) and (5) is a

standby emergency system, containing enough chemical to remove one day's production of CO_2. This system is intended for use in the event repairs should be required to the molecular sieve system.

ION BALANCE

It has not as yet been definitely established that ion control will be required. There is evidence to indicate that if the inspired air contains more positive small ions than negative small ions some adverse physiological and psychological effects occur. If it is determined that air ionization is required, it probably will be accomplished by one or more tritium sources, each of which produces in the order of a billion ions/second. The positive ions will be attracted to a negative electrode and neutralized, and the negative ions will be discharged into the cabin.

PARTICULATE MATTER AND BACTERIA CONTROL

Particulate matter and bacteria will probably both be removed by a static particulate filter, although specialized electrostatic filters are still under investigation. A relatively high-efficiency filter is required to remove the particle sizes which are, physiologically, the most dangerous. Both the static particulate and the electrostatic type filters will remove, with a high efficiency, both dust-borne and droplet nuclei bacteria.

ATMOSPHERIC SENDING AND ELECTRICAL CONTROL SYSTEM

As a minimum, the wall temperature, atmospheric temperature, cabin wall temperature, pO_2, pH_2O, pCO_2, total pressure and several atmospheric flow rates will be sensed and the data either displayed to the crew or telemetered to Earth, or both. Some information will be fed back to the gaseous control system to regulate the operation of specific subsystems. At present, it appears that a time-of-flight mass spectrometer will be used to determine the gaseous composition of the atmosphere. In

addition to providing the basic information required, the mass spectrometer under consideration can be used to provide broad-spectra atmospheric data which can be telemetered back to Earth for analysis by a trained spectroscopist. Polarographic pO_2 and pCO_2 sensors are being considered as a back-up to the mass spectrometer.

FIRE DETECTION AND CONTROL SYSTEM

The fire detection and control system has not been defined completely as yet. It is probable that the heart of the detection system will consist of a thermistor temperature-detection system. It is expected that a fire extinguisher will be provided for both the command and the mission modules. Various types of fire extinguishing materials are being investigated to determine their suitability for use in a closed system. Procedures are being established for use in the event of fire. These include such things as stop all blowers, de-energize all nonessential electrical equipment, stop O_2 supply, etc. The lack of convection can be a powerful fire-fighting tool. The oxygen molecules required to support the conflagration can, without convection, only reach the point of combustion by diffusion through the inert sphere of gas, consisting of the diluent and the products of combustion, surrounding the point of combustion.

LEAK DETECTION SYSTEM

The existence of small leaks can be determined by comparing present and past diluent consumption rates over a period of time of sufficient duration to cancel out the effect of changes in temperature, etc. Although providing an indication of the size of leak, no indication of location is available. An alternate means can provide both an indication of magnitude and general location. This concept uses ionization-type probes located about the exterior surface of the pressure hull. If a leak exists in the vicinity of the probe, the gas is ionized, thereby "activating" the particular probe. Since the probe location is known, the general area of the leak is thus revealed.

EFFECTS OF MODULARIZATION

The existence of two modules requires that consideration be given to the location of the atmospheric control hardware both for normal operating conditions and in the event of the loss of the pressure integrity of either or both modules. For analysis, three cases are assumed:

(1) The normal case, where the vehicle successfully completes the mission, as planned, with both the command and the mission modules pressurized. The crew will all be in the command module for powered flight and re-entry. The balance of the time, one man will be in the mission module except, perhaps, for short periods of time when thrust is applied, either for course correction or for injection into, or ejection out of, the lunar orbit.

(2) The mission module develops an irreparable leak just at the point of no-prior return. For the purpose of this discussion, this point is assumed to be passed three days after lift-off. The return flight is assumed to take an additional half-day to the Moon and 3-1/2 days from the Moon to the Earth, giving a total of four days without pressure in the mission module. This assumption of pressure loss at the point of no-prior return gives the longest emergency flight time possible.

(3) The command module irreparably loses its ability to hold pressure, just as the point of no-prior return is passed.

For the first case, it is assumed that the command module hatch will remain open. Cooling, CO_2 removal, water vapor control, and oxygen replenishment can all be accomplished in the command module. Circulation between the modules can be accomplished as follows: A hinged support will be mounted in the hatchway of the mission module which carries two blowers, one of which blows a jet into the mission module and the other blows a jet into the command module. A circulation rate of about 100

CFM is all that is required — and should easily be achieved — to maintain the temperature, CO_2, and humidity levels within the mission module at approximately the same value as in the command module.

Should case (2) occur, i.e. the mission module irreparably loses pressure, then all three men will remain in the command module. All of the environmental control equipment is located with them and will continue to operate to maintain a "shirt-sleeve" environment within the command module. Adequate gas reserves are carried to permit several visits — in a pressure suit — to the mission module, should this be required. Before the hatch is opened between modules, it will be necessary to equalize the pressure within the modules. This will require the loss of something between one-half and full atmospheric content of the command module, depending on the size hole in the mission module.

Should case number (3) occur, i.e. the command module irreparably loses its pressure integrity, then several choices are available. Either all three crew members can remain in the command module in their secondary pressurization system, or two can remain in the command module and one in the mission module, with the frequency of transfer back and forth depending upon how much of the gas reserves are used up (by module pressure equilibration and from leaky pressure suits) with each transferral and on the amount of gas reserves available. In addition to the question of gas reserves, other factors which will affect the decision as to whether or not a crew member will stay in the mission module are the habitability of the secondary pressurization system, (i.e. will cocoons or pressure suits be used, what provision is available for waste disposal, what is the availability and palatability of food and water) and the importance of the tasks which possibly can only be performed in the mission module (i.e. navigation, etc.). If the visitor to the mission module expects to stay for only a short time, it will not pay to pressurize the mission module while he is there. Instead, he will be supplied via an umbilical from the oxygen supply system to his pressure suit. If it is necessary or desirable for him to remain in the mission module for a long time, the oxygen required to pressurize the module can keep him for at least four hours without any additional supply. In addition, for this period of time, the CO_2 concentration will not build up to a dangerous level. Therefore, assuming the module is thermally controlled, no

additional atmospheric control equipment – except for the O_2 line required for pressurization – will be needed in the mission module. If it appears desirable to keep one man in the mission module for periods longer than this, he can use the portable lithium hydroxide emergency system which has adequate capacity.

If, in addition to the problem of loss of pressure integrity of the command module, the pressure integrity of the environmental control system – which now is supplying conditioned atmosphere to the secondary pressurization system – is lost, then individual conditioners will be available for two of the three cocoons, or full-pressure suits. These conditioners will each have adequate capacity to carry the metabolic thermal and gaseous load of a crew member for several hours, giving adequate time (barring a complete catastrophe) for repair of the environmental control system.

In summary, all of the environmental control equipment will be located within the command module except for a small-capacity heat exchanger and a part of the gas supply, which will be located within the mission module.

Small-diameter gas supply lines, plus the appropriate valving, will be available so that any or all of the gas supply can be discharged into either module as desired.

12. OXYGEN SUPPLY

At present, there exist in the industry several feasible methods of supplying the oxygen required aboard a space vehicle such as APOLLO. These are as follows:

1. Stored oxygen
   a. In the gaseous state
   b. In the liquid state
   c. In the supercritical state
2. Chemical reaction
   a. Reaction of potassium superoxide (KO_2) with carbon dioxide (CO_2) and water (H_2O).
   b. Decomposition of hydrogen peroxide (H_2O_2).

Regenerative methods of oxygen supply, such as electrolysis of water, decomposition of carbon dioxide, and controlled growth of algae were not considered for this project because of the large weight and volume penalties which are characteristic of these systems.

In considering the oxygen requirements for APOLLO, it was decided that a primary oxygen storage system and a spare storage system be provided. The primary storage is calculated to suppy metabolic needs and to compensate for the oxygen leaked from the cabin. The spare supply is included to supplement the primary supply, in the event the crew requires more oxygen than is expected, the vehicle leaks more than is expected, or it becomes necessary to purge the cabin atmosphere to remove noxious or toxic gases.

A quantitative breakdown of the primary and spare storages is given in the following table.

Use	Weight of Oxygen	
	Primary	Spare
metabolic	1.6 (3) 14 = 67.2	0.2 (67.2) = 13.4
leakage	15.5	--
purges(2)	--	12.6
Totals	82.7 pounds	26.1 pounds
	Grand Total = 108.8 pounds	

Weight and volume trade-offs were performed to compare high pressure, supercritical and liquid storage of oxygen, the reaction of potassium superoxide and the decomposition of hydrogen peroxide. Graphs of "primary oxygen storage weight" and "primary oxygen storage volume" versus "flight time" are included. See Figures XI-12-3 and XI-12-4.

HIGH PRESSURE STORAGE

Oxygen storage at high pressure has the advantage that it is capable of being dispensed, under zero-g conditions, with no special hardware. Storage in three types of pressure vessels is compared: steel, filament-wound fiberglass, and titanium tanks. As can be seen from Figure XI-12-1, a considerable spread is indicated for the fiberglass tanks. The data used in preparing this estimate was received from several sources(1). The spread in the data, along with a history of trouble with fiberglass pressure vessels indicates that the art of design and manufacture has not been perfected. However, this is not adequate reason for not considering fiberglass tanks, as advancements are being made in the state-of-the-art.

When high-pressure storage is considered, a trade-off can be made between weight and volume. Up to about 300 atmospheres, the container weight for a given weight of oxygen is almost independent of pressure. However, above this pressure, the fairly rapid increase in the compressibility factor increases the weight of the container, since the container weight is proportional to the compressibility factor. This is shown in Figure XI-12-1. Figure XI-12-2 compares volume and compressibility factor as a function of the container pressure. The 7500 psi (500 atmosphere) pressure was selected for comparison with the 4500 psi (300 atmosphere) pressure because, at 7500 psi the product of pressure and volume is minimized(2), and for this reason it has been used in past systems.

References:

(1) Personal communications from Lamtex Industries and Walter Kidde, Inc.

(2) WADD Technical Report 60-33

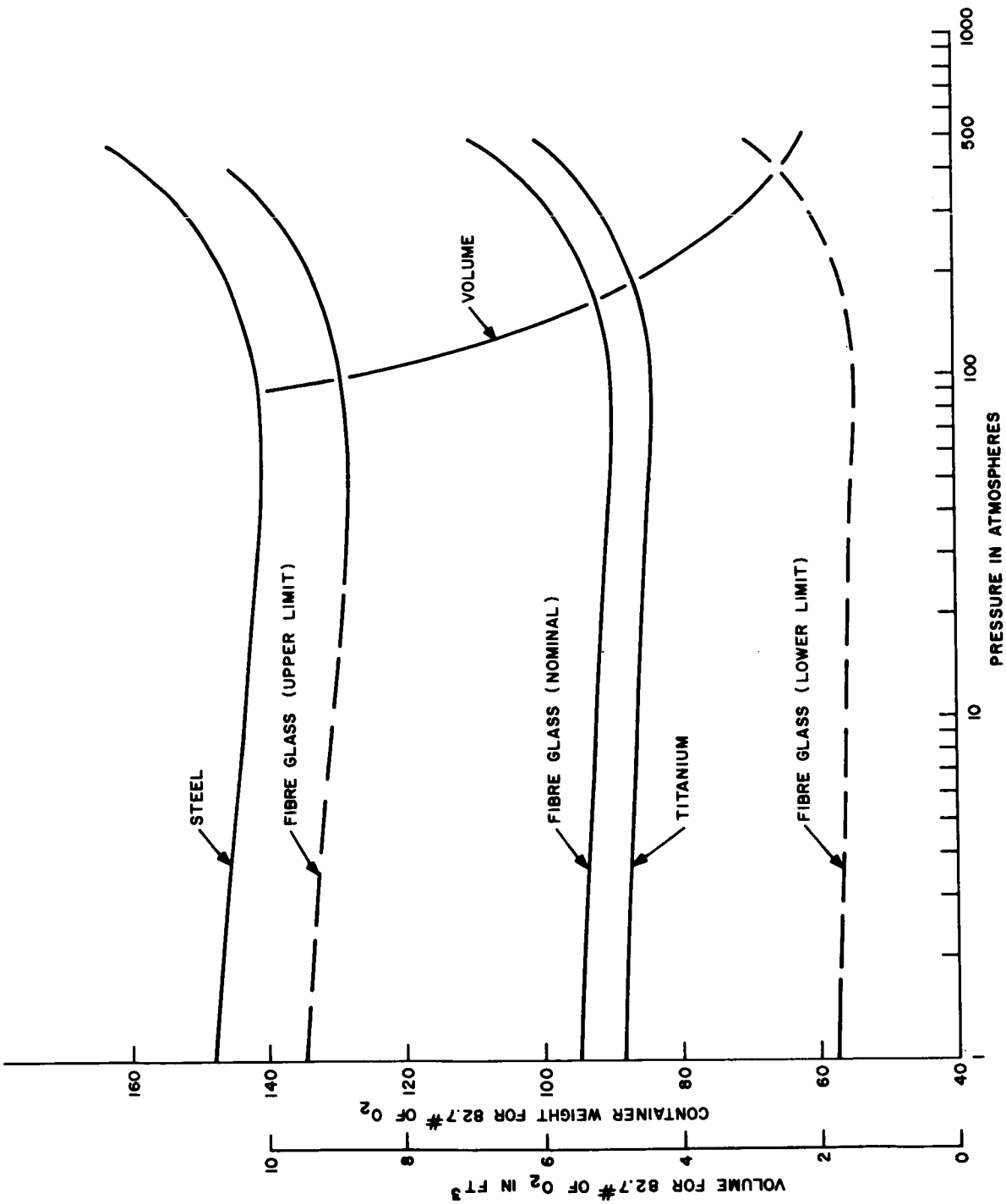


Figure XI-12-1. Container weight and volume versus oxygen pressure at temperature (T) = 68 F

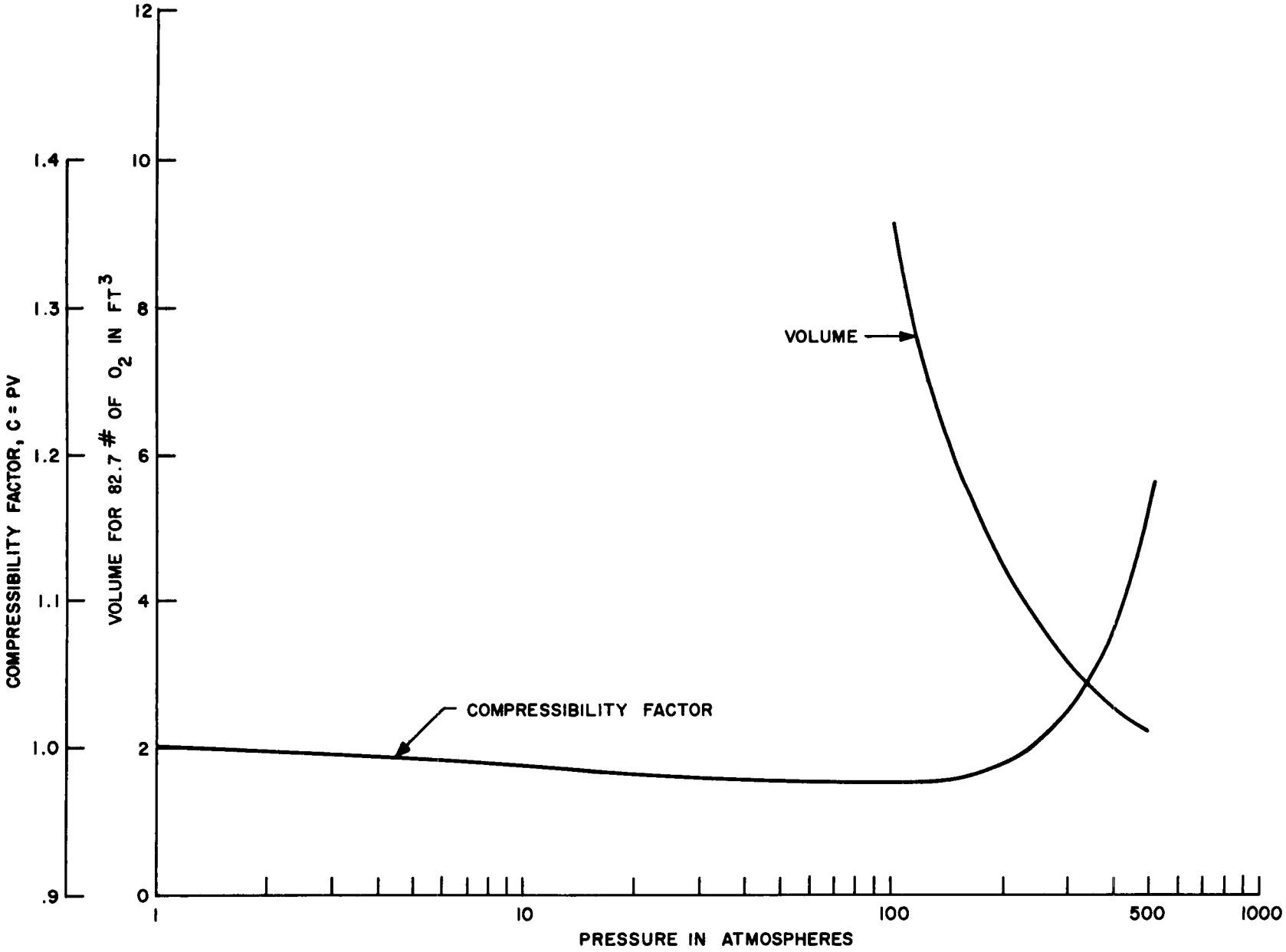


Figure XI-12-2. Compressibility factor and volume versus oxygen pressure at temperature (T) = 68 F

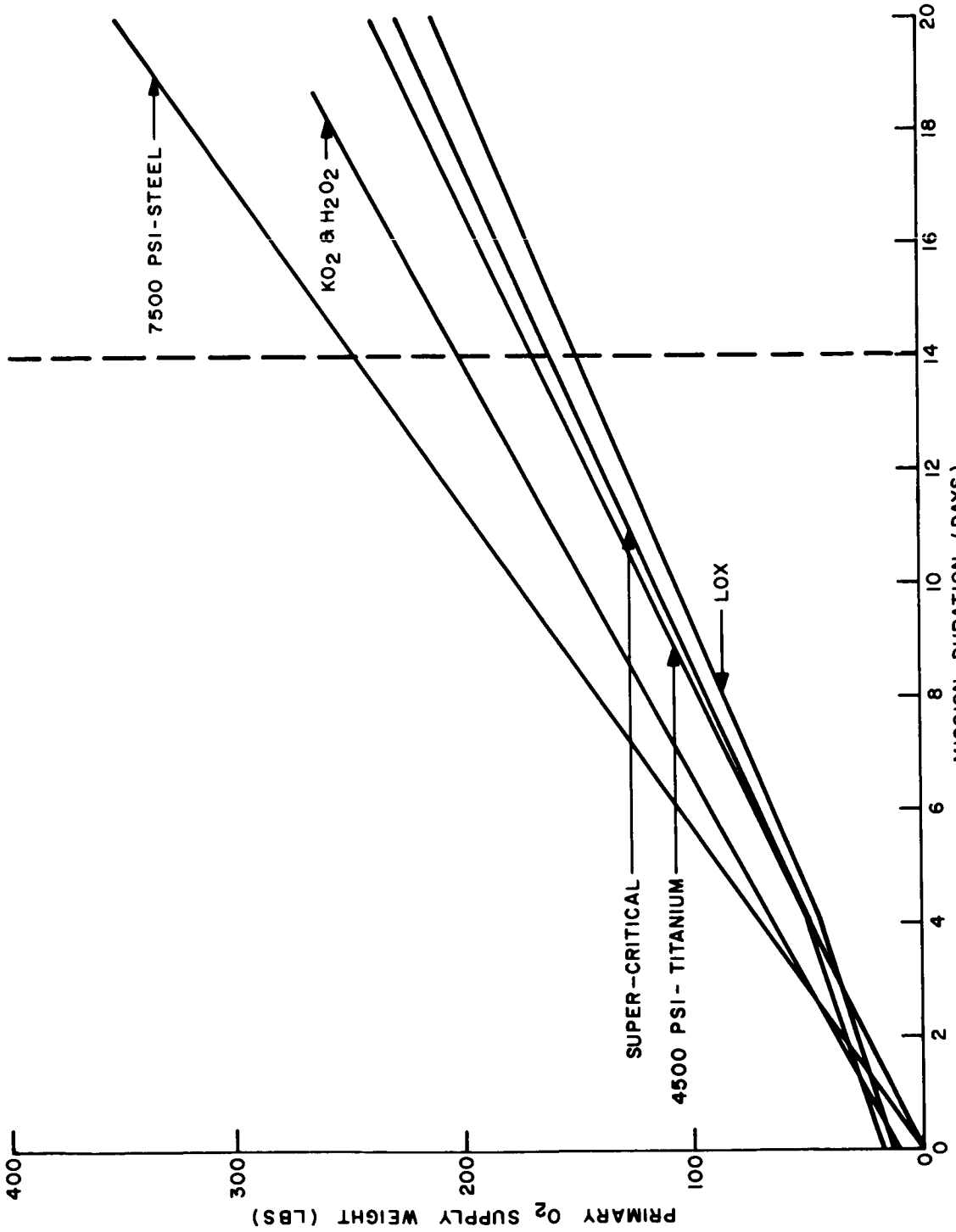


Figure XI-12-3. Primary oxygen supply system weight versus mission duration

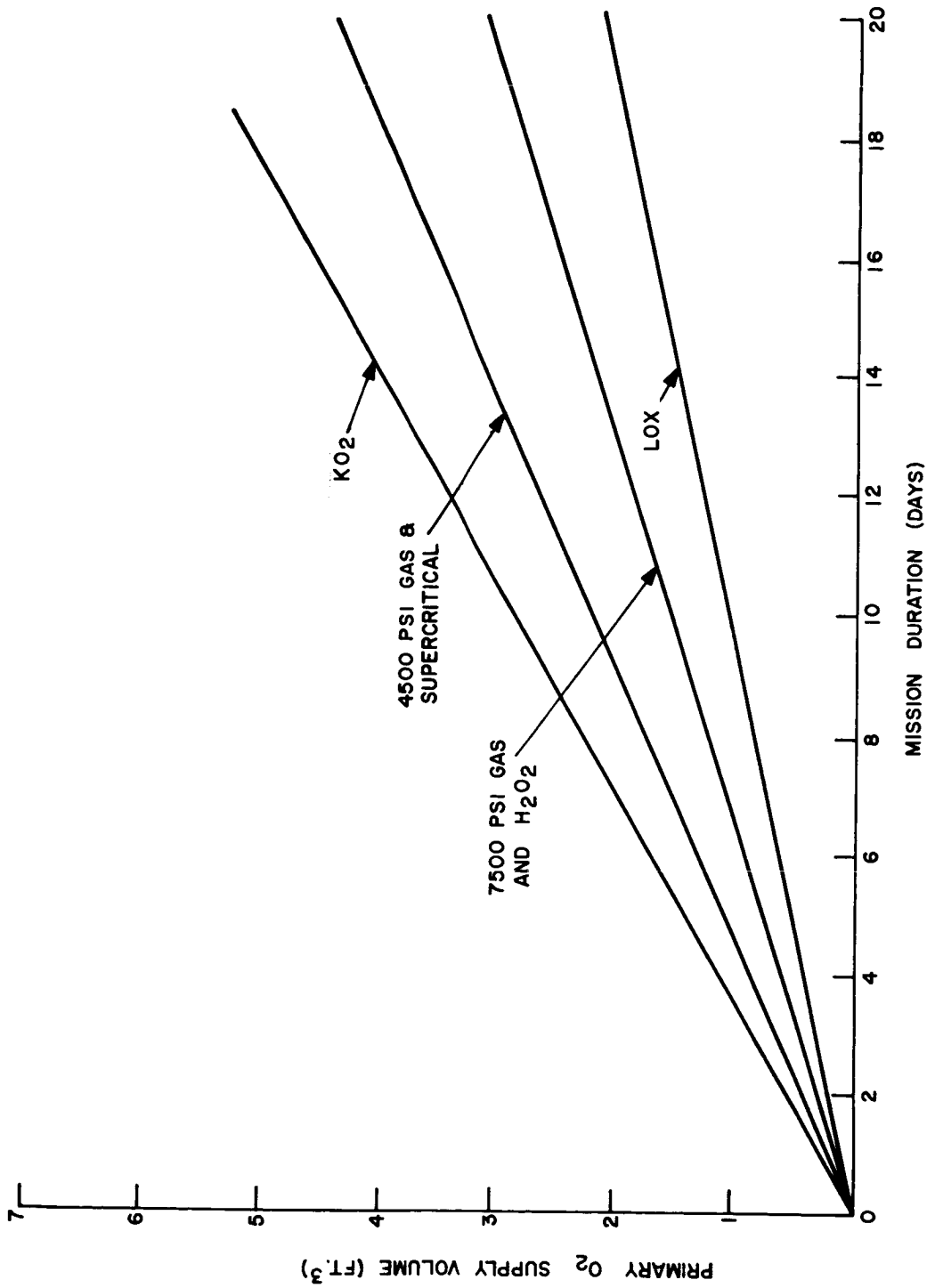


Figure XI-12-4. Primary oxygen supply system volume versus mission duration

LIQUID STORAGE

Although the problem of phase separation in a zero-g environment between liquid and gas has not been completely solved, there is reason for believing that it will be solved in time for use in the APOLLO vehicle. Several companies have been studying the problem, including the General Electric Company which has proposed two different liquid oxygen converter designs for use in zero-g. One design makes use of a physical barrier to maintain phase separation under all external conditions. Positioning of the barrier is accomplished during filling of the Dewar. A selective valve situated in the barrier permits gas to pass to one side of the barrier, while the liquid is confined to the opposite side. A second converter design stores the liquid in a network of capillaries which, assisted by a gaseous head pressure, effects a separation of liquid and gas at all Dewar altitudes and in all gravitational fields. The weight of the liquid oxygen storage system has been calculated at 0.8 pounds per pound of oxygen, giving a total system weight, including the oxygen, of 1.8 pounds per pound. This weight applies to converters holding more than approximately 25 pounds of oxygen. For smaller-size converters, the system weight will be slightly greater on a pound-per-pound basis.

SUPERCRITICAL STORAGE

Supercritical storage of oxygen in this discussion means the cryogenic storage of oxygen at low temperatures and moderate pressures. The container weight for supercritical storage will be about 0.96 lbs/lb O_2. The storage of supercritical oxygen does not present the problems connected with liquid storage because the oxygen is still gaseous and, thus, no phase change or phase separation is needed. Zero-g operation is, therefore, not a problem. Development is required, however, to regulate the heat leak into the Dewar to maintain the temperature and pressure above the saturated vapor phase line as the oxygen supply is used. Otherwise the oxygen would liquefy.

HYDROGEN PEROXIDE STORAGE

Calculations for the hydrogen peroxide system assumed a solution of 90 percent peroxide by weight. Operation of a hydrogen peroxide system requires the use of a catalyst to promote decomposition and a cooling system to reject the heat of decomposition and to condense the generated water vapor. Sixty pounds and one cubic foot has been subtracted from the hydrogen peroxide system weight because this amount of water is required for crew support. A weight of 10 pounds was assumed for both of these components. In addition, the container weight for peroxide was assumed to be 0.3 pounds for each pound of hydrogen peroxide solution stored. System volume was considered to closely approximate the volume of the hydrogen peroxide, except, as indicated above, the one cubic foot credit for the water obtained.

POTASSIUM SUPEROXIDE STORAGE

The theoretical yield of the potassium superoxide system is one pound of oxygen for every 3.0 pounds of superoxide stored. The KO_2 absorbs CO_2 in the process of liberating oxygen. Assuming that, through controlled formation of the bicarbonate, the R.Q. of the system and men can be matched, we can subtract about 90 pounds and 2.3 cubic feet (the weight and volume of the lightest CO_2 removal system) from the KO_2 system, to account for the fact that it also absorbs CO_2. See Figure XI-14-1. This has been done in Figure XI-12-3. Ten pounds has been assumed for the required valves and controls. The volume of a superoxide system is assumed equal to the volume of required chemical.

TRADE-OFF FACTORS

The system weights for the various methods of primary oxygen supply are given in Figure XI-12-3. The volumes corresponding to the above systems are given in Figure XI-12-4. A titanium tank, operating at 4500 psi, was selected for the comparison of gaseous supply because this represents the lightest proven method for gaseous storage (refer to Figure XI-12-1). A fiberglass tank may ultimately be lighter in weight, but

this remains to be determined in development. A steel tank at 7500 psi is also compared, because, as mentioned previously, at this pressure the product of pressure times volume reaches a minimum. Steel was chosen for its relative ease of fabrication, and proven ability for use at this high pressure.

Examination of Figure XI-12-3 indicates that the liquid oxygen converter affords the least weight penalty for oxygen storage in APOLLO. A supercritical oxygen system weighs only 13 pounds more. Volume of the supercritical system (Figure XI-12-4), however, is more than two times the volume of an equivalent liquid oxygen system. Successful operation of the supercritical system is dependent primarily on balancing the heat leaked into the container against the heat required to maintain the oxygen above its critical pressure (50.1 atmospheres) and critical temperature (154.7 K). A decrease in temperature or pressure will, in the early phase of usage, result in liquefaction of the oxygen; excessive heat gain, on the other hand, will result in too high a pressure.

Operation of the liquid oxygen system does not depend upon a balanced heat leak. Passive heat transfer to the converter is not essential to the operation of the system, nor does it present a hazard. During normal operation, enough heat will be transferred passively to the oxygen discharge coil to vaporize all the liquid required to sustain metabolism and anticipated leakage.

A titanium pressure vessel for oxygen storage at 4500 psi is a good alternate choice. It weighs 10 lbs more than the supercritical system and has the same volume. This 10 lbs may, in fact, become nonexistent once a supercritical system is developed and the power required to regulate the supercritical pressure and temperature becomes known, i.e. the difference in weight at this point is not significant. However, the titanium pressure vessel weighs 23 lbs more than LOX system and occupies two times the volume, or 1.5 cu ft more. This additional weight and volume penalty is significant and tentatively favors the LOX system. The gaseous system remains, however, the simplest, and thus inherently the most reliable of any. If major problems occur in the development of the liquid converter; high pressure storage in titanium tanks — or possibly fiberglass tanks, depending on the state-of-the-art—can be reverted to without a prohibitive weight penalty.

13. DILUENT SUPPLY

DILUENT SUPPLY STORAGE

In Section XI-10, "Atmosphere Selection", nitrogen was selected as the diluent gas. In addition, it was determined that the total cabin pressure will be 360 mm Hg and the nitrogen partial pressure will be 167 mm Hg.

The amount of nitrogen required to be stored aboard the vehicle will be the sum of the nitrogen required to make up that lost through leakage, plus the nitrogen required to repressurize the cabin after purging. It should be noted that a cabin purge is not a programmed part of the mission, but will arise from emergencies only, such as fire or toxic gas control failure. The nitrogen leakage, from Figure XI-10-3, is 12.5 lbs for the total duration of 14 days, assuming a leak rate of 1000 cc/min at cabin pressure. The amount of nitrogen required to repressurize the 300 cu ft cabin to a partial pressure of 167 mm Hg is 4.75 lbs. The total amount of nitrogen required, assuming two purges per mission, is then 12.5 + 9.5 = 22.0 lbs. A safety factor need not be applied to this total, as it is estimated that the leak rate of 1000 cc/min is an upper limit which, will not be exceeded.

NITROGEN STORAGE

Nitrogen may be stored as a liquid or as a gas. Comparisons will be made below of the trade-off factors of weight and volume associated with gas storage in a steel, fiberglass, or titanium container; cryogenic (liquid) storage; and cryogenic (supercritical) gas storage.

Gas

Two factors are of importance in the gaseous storage of nitrogen — weight and volume. Figures XI-13-1 and XI-13-2 show the compressibility factor, volume, and container weight as a function of the nitrogen pressure of the container. As shown, from a volume viewpoint, the operating pressure of the container should be as high as

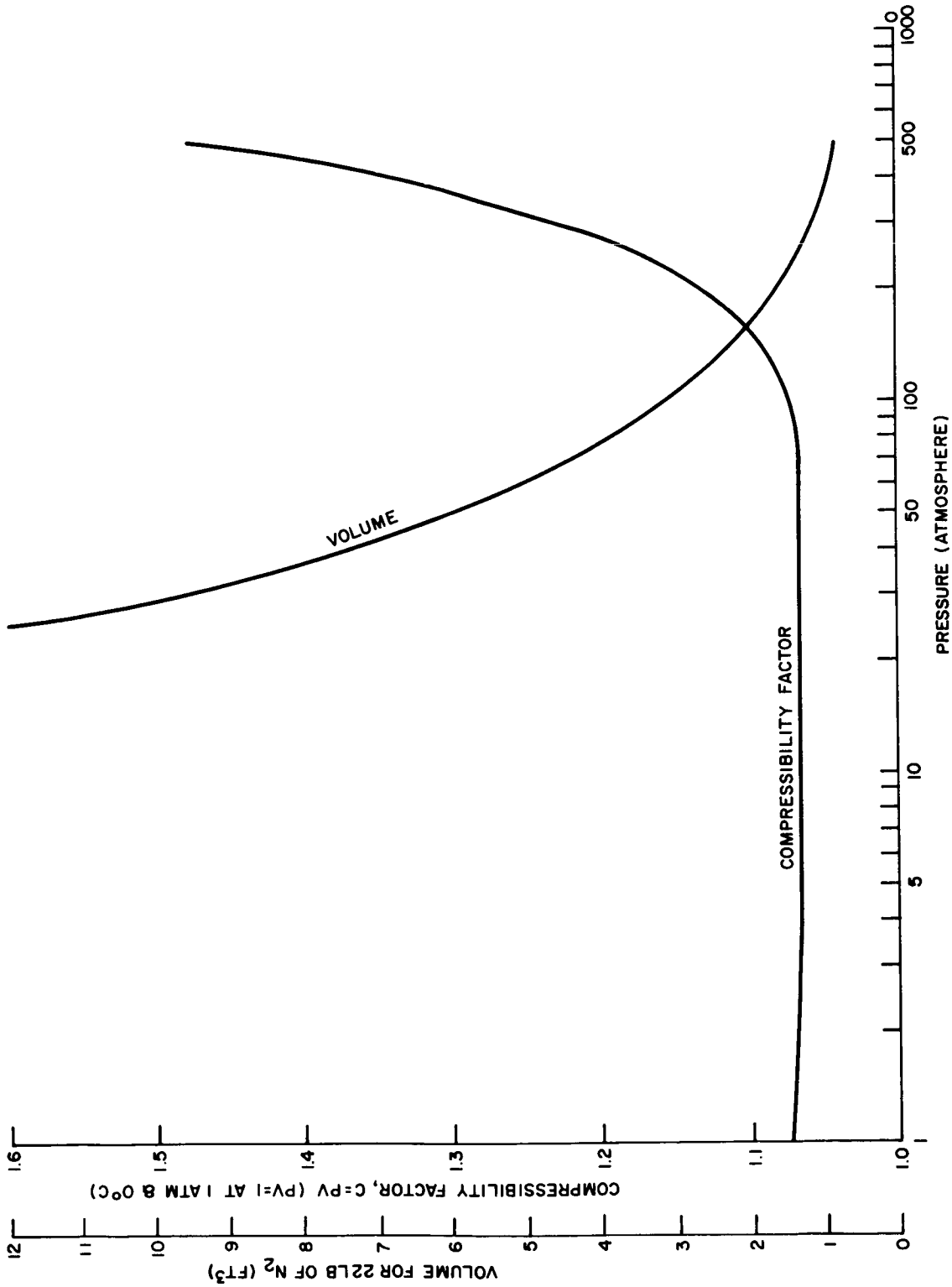


Figure XI-13-1. Compressibility factor and volume versus nitrogen pressure (T = 68F)

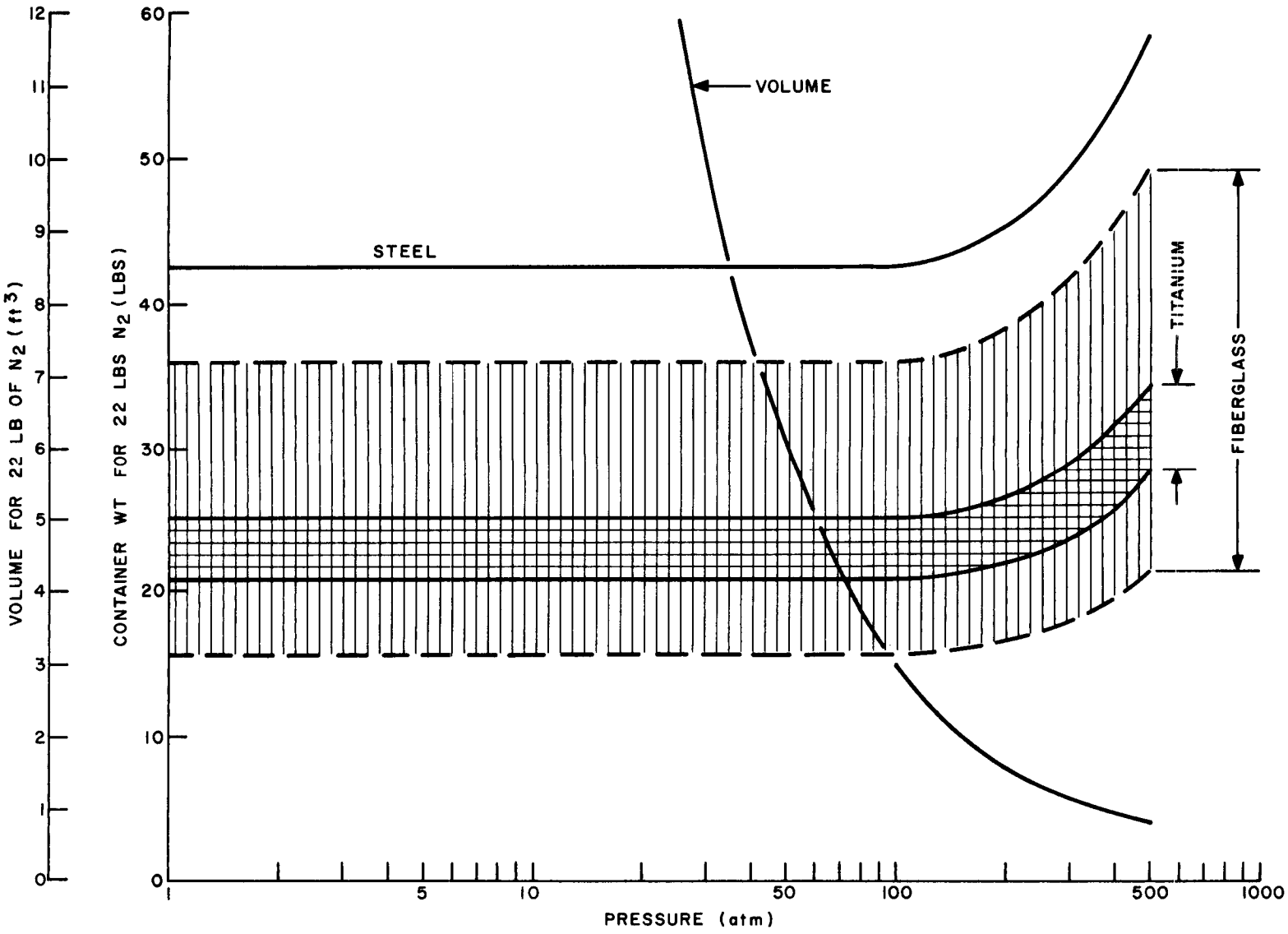


Figure XI-13-2. Container weight and volume versus nitrogen pressure (T = 68F)

possible, since this results in the lowest volume. From a weight standpoint, however, the container weight will be proportional to the compressibility factor. This means that the container weight for a fixed amount of nitrogen will be constant with pressure up to about 100 atmospheres, above which the container weight will increase. From Figure XI-13-2, it can be seen that the container weight increase is relatively small between 100 and 200 atmospheres while the volume is still decreasing rapidly. Above 200 atmospheres, the container weight increases much more rapidly while the resultant decrease in volume is much lower. In the APOLLO vehicle, it is considered that the decrease in volume obtained by increasing the pressure above 200 atmospheres is not sufficient to warrant the additional weight penalty incurred. Therefore, the design pressure for gaseous nitrogen storage will be selected at 3000 psi (204 atmospheres).

Three materials are considered for pressure vessels — steel, titanium, and fiberglass. Of these, steel has the highest weight-to-strength ratio and would consequently make the heaviest container. However, steel has certain advantages in relative ease of fabrication, familiarity, and use. At 3000 psi, a steel container designed to store 22 lbs. of N_2 would weigh 45.3 lbs. Thus, the storage supply weight would be 45.3 + 22 = 67.3 lbs. At this pressure, the storage supply volume would be 1.55 cu ft.

The weight factor assigned to the titanium tank is 2.2 lbs of N_2 plus tank per lb of N_2 delivered. This number is based on existing designs, close to the design requirements for the APOLLO tank (i.e. operating pressure of about 3000 psi, burst pressure over 6000 psi) which have been tested by the General Electric Company.

The material used in the fabrication of these tanks is a titanium alloy containing 6 percent aluminum and 4 percent vanadium. Heat treated, it has an ultimate tensile strength of 160,000 to 170,000 psi. Because of the difference in density between steel and titanium, this is equivalent to steel heat treated to 280,000 to 300,000 psi UTS. (The steel bottle used for the comparison above is fabricated of AMS 6434 heat treated to 180,000 to 200,000 psi UTS.) Both Airite Products, Inc and Menasco Manufacturing Company, two leading manufacturers of titanium pressure vessels, make their tanks of this alloy.

An all-beta alloy, B-120 VCA, developed by the Crucible Steel Company shows promise for pressure vessel applications. This alloy has an ultimate tensile strength exceeding 190,000 psi, and shows excellent promise of reducing the weight factor to 2.0.

In the case of the filament-wound fiberglass pressure vessel, difficulty is encountered in assigning a weight factor (lbs/lb N_2) for comparison. Data from several reputable sources indicate that the weight of container plus N_2 for a 5000 psi fiberglass vessel would be from 1.83 lbs/lb N_2 to 2.95 lbs/lb N_2. A 3150 psi fiberglass tank was tested by G.E. which had a weight of 2.3 lbs/lb N_2. Although these tanks did not completely pass the testing to which they were subjected; the cause, and it is believed the cure, for the failure is known. A development program is necessary to further refine the actual weight between the limits of 1.83 to 2.95 lbs/lb. The volume, as for the other tanks, is 1.55 ft^3.

A nitrogen supply system utilizing gaseous nitrogen at 3000 psi would be the simplest and, probably, the most reliable. No new techniques are required and zero-g operation is assured.

Liquid

One cubic foot of liquid nitrogen will supply 641 cu ft of STP nitrogen gas. The volume of liquid required is then $\frac{(22)}{(0.077)(641)} = 0.445$ cu ft. The liquid nitrogen converter (based on a similar LOX converter) will weigh 1.1 lbs per lb of N_2. Therefore, the total storage weight would be (2.1)(22) = 46.2 lbs.

The design problems in a liquid nitrogen converter would be much more difficult than the gaseous storage system considered above. This stems from the conversion difficulties encountered in a zero-g environment which does not provide the easy separation of the liquid and gas phases. One design which shows merit utilizes a physical barrier to separate the phases. It will suffice to say that the design problems, though difficult, are not insurmountable.

Supercritical

Supercritical storage of nitrogen in this discussion means the cryogneic storage of gaseous nitrogen at low temperatures and moderate pressures, close to the critical point. The container weight for supercritical storage is calculated as follows:

Wt. of dewar and mounting (for 25 lb N_2 capacity)

= 10 lbs. (based on LOX converter, good for 300 psia)

$\therefore$ container wt. $= \frac{10}{25} = 0.4$ lbs/lb N_2

Now, volume is proportional to dia^3

and, surface area is proportional to dia^2

and, container weight is proportional to surface area

$\therefore$ container weight is proportional to vol$^{2/3}$.

The density of liquid $N_2 = 0.69$ gm/cc and the density of supercritical $N_2 = 0.31$ gm/cc (at 492 psia).

Then container weight $= 0.4 \frac{493}{300} \left(\frac{.69}{.31}\right)^{2/3} = 1.13$ lbs/lb N_2.

The volume occupied by the $N_2 = \frac{(22)}{(0.31)(62.4)} = 1.14$ ft^3

and the storage weight is $(2.13)(22) = 46.8$ lbs.

The storage of supercritical nitrogen does not present the problems connected with liquid storage because the nitrogen is still gaseous and, thus, no phase-change or phase separation is needed. Zero-g operation is, therefore, not a problem. Development is required, however, to regulate the heat leak into the dewar to maintain the temperature and pressure above the critical point, thus preventing liquefication.

SUMMARY

High pressure storage has been tentatively selected for the APOLLO vehicle because of the simplicity of the system. With future developments anticipated in the state of the art of both titanium and filament wound fiberglass containers the weight should be no greater than that of a liquid supply system. However if, as the study continues, the relatively high volume requirement becomes more significant than the anticipated heat leak or conversion problems, then either the supercritical or liquid storage system will be selected. Figure XI-13-3 compares the weight and volume of the various methods of storage as a function of time.

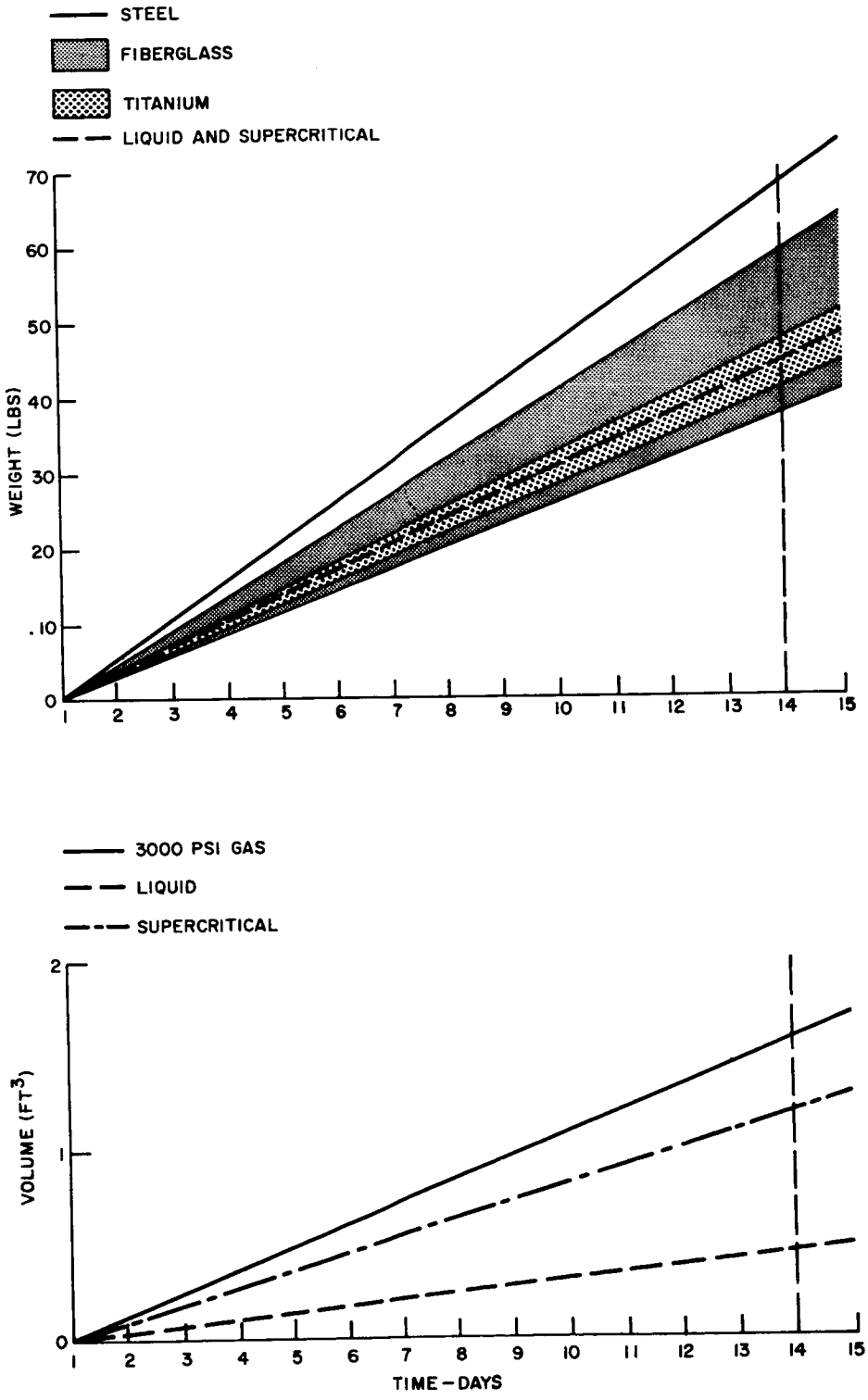


Figure XI-13-3. Nitrogen storage versus time

14. CARBON DIOXIDE REMOVAL SYSTEM

As previously mentioned, the partial pressure of CO_2 in the cabin enclosure must not exceed 8 mm Hg. The problem is to select the most desirable method to separate and remove the carbon dioxide from the air. The first phase is to separate the feasible methods from the large number of possibilities. The following ways of CO_2 separation were considered feasible.

(1) Freeze-out System - A heat exchanger surface is cooled to a sufficiently low temperature to freeze out the CO_2.

(2) Potassium Superoxide - The air is exposed to KO_2 chemical, resulting in the absorption of CO_2 and the production of O_2.

(3) Lithium Hydroxide - Air is circulated through a LiOH system, with the CO_2 being absorbed chemically.

(4) Molecular Sieve - The solid absorbent molecular sieve is used to selectively absorb the CO_2 gas from the air. The vacuum of space is used to reactivate the molecular sieve in a cycling type system.

An objective analytical study was used to compare the performance of the four methods. For an accurate comparison, the systems were modified to bring out each system's full potential. Numerical comparisons were made with calculations and trade off curves. Figure XI-4-1, which assumes a power penalty of 400 lb/kw, indicates that the molecular sieve system is about half the weight of the second best system. Additionally, the future growth potential and low volume requirements of the molecular sieve method of CO_2 control are important factors in its favor.

The functional schematic of the CO_2 removal system, shown on Figure XI-4-2, is one possible molecular sieve system which includes four solid absorbent beds, two of silica gel and two of molecular sieve. The system operation is explained in Section XI-11.

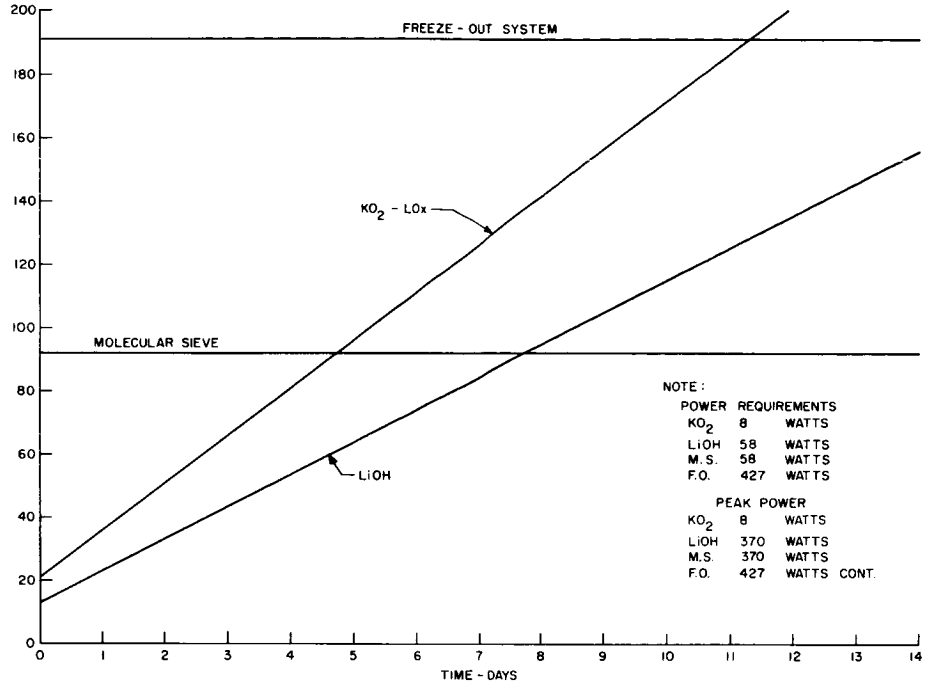


Figure XI-14-1. CO_2 removal system

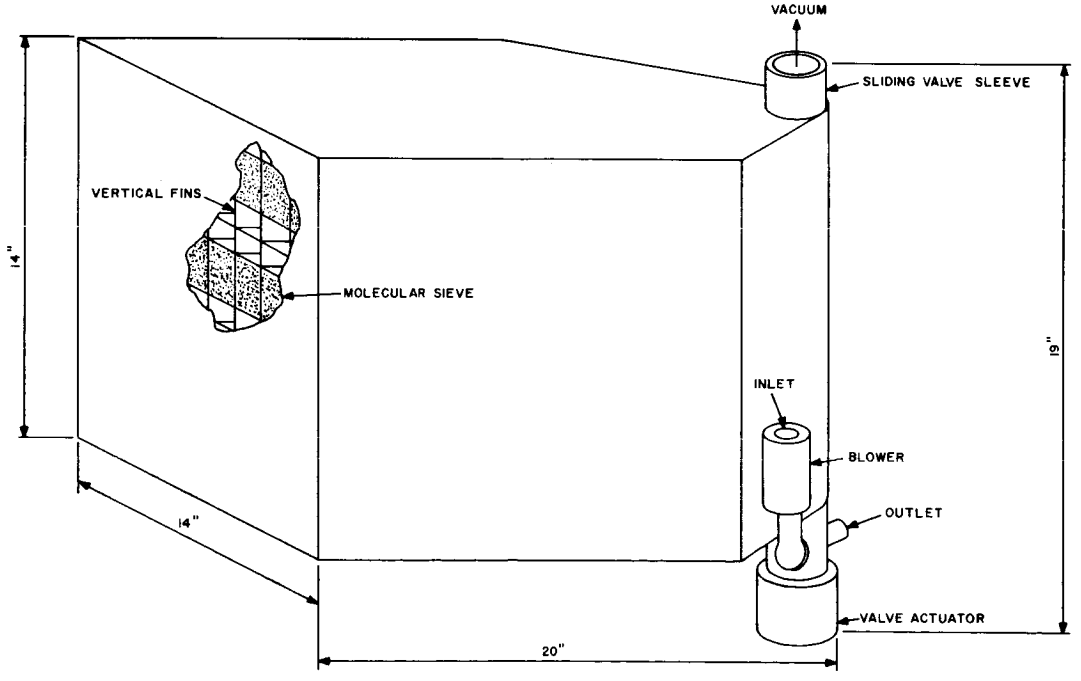


Figure XI-14-2. CO_2 adsorber

This system includes a large amount of tubing and complicated valving. A simplified hardware system is shown in Figure XI-14-2. A sliding valve arrangement takes the place of the three valves shown in the schematic. See Figure XI-14-3. The valve is a vacuum-tight "O" ring sealed device that requires little power to operate. If valve failure occurs, leakage of cabin air to space cannot occur due to a fail-closed feature. A manual override allows emergency use. The large amount of finning in the beds causes inter-bed heat transfer of the heats of adsorption and desorption. This greatly reduces the amount of electrical heat required for desorption of the water vapor and increases the over-all efficiency of the system. However, provision has been made for heating the silica gel beds for 15 minutes every 5 cycles to insure that water poisoning of the molecular sieve bed cannot occur.

During regeneration, the molecular sieve bed is exposed to vacuum on both sides. This results in a very short flow path for the CO_2 to reach the vacuum, thereby insuring a high percent regeneration capability.

Because it appeared, early in the study, that the molecular sieve system would be the lightest CO_2 control system for the APOLLO vehicle, the Hamilton Standard Corporation was asked to perform a parallel, but independent study. The over-all weight and power, which they calculated for their molecular sieve system, closely confirms the conclusions of the General Electric Company.

CONFIDENTIAL

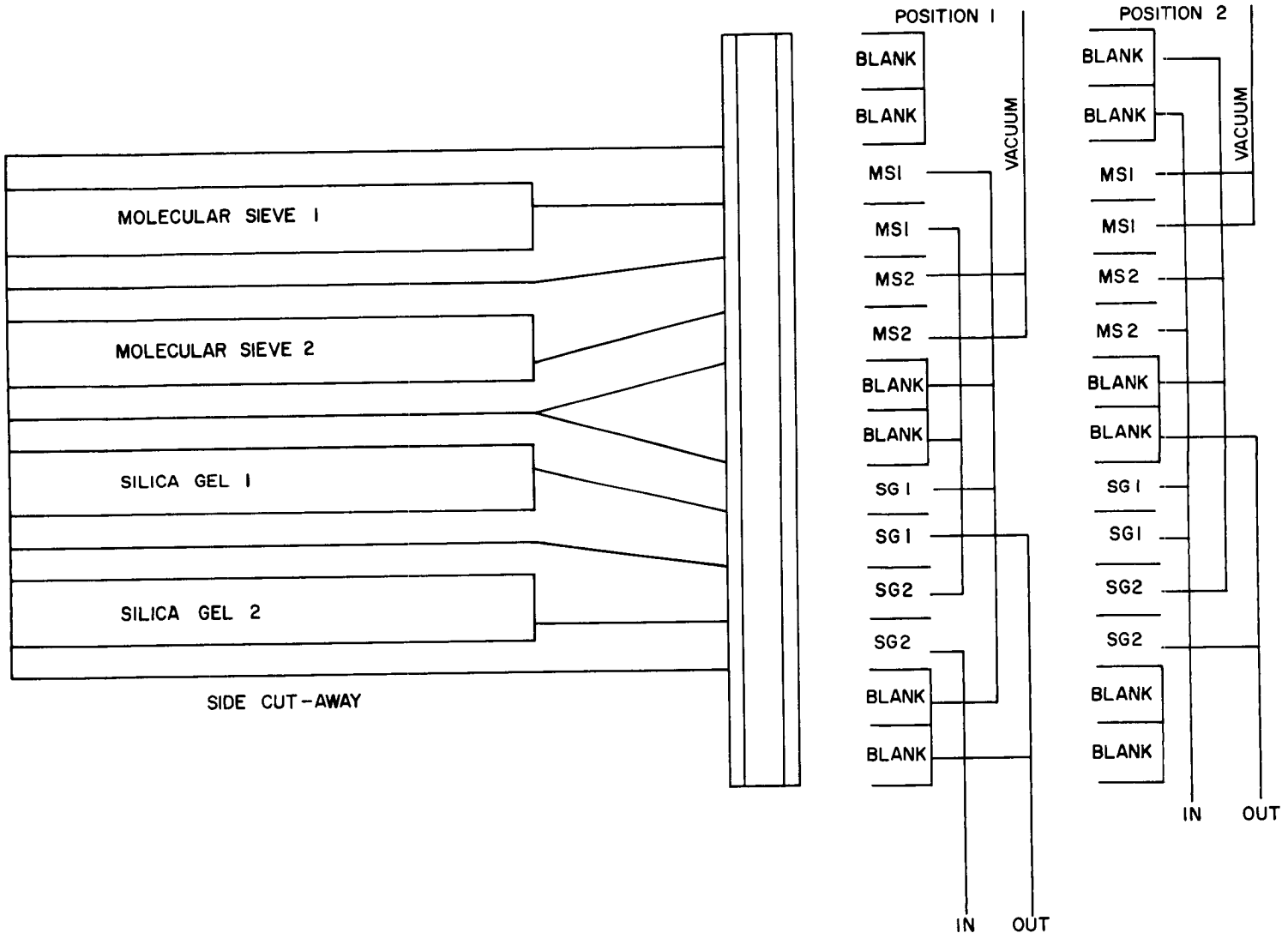


Figure XI-14-3. Functional schematic of CO_2 adsorber sliding valve

CONFIDENTIAL

15. WATER VAPOR CONTROL

The humidity of the cabin atmosphere is to be kept between 5 and 15 mm Hg partial pressure. Since it will be initially at or near this level, the H_2O removed must be equal to the moisture added to the atmosphere in the form of perspiration and respiration. This is estimated at 1.87 lb/man/day (this is probably a low value but, because of the type trade-off made, it is a conservative one). This H_2O, if removed from air which is filtered of bacteria, is acceptable for use as drinking or wash water. Therefore, any system which does not save this H_2O for future consumption must have the weight of the water lost included in its system weight.

Silica gel, lithium chloride and condensation on the heat exchanger are evaluated as humidity control means and trade-off data is presented in Figure XI-15-1. This data clearly shows the advantage of the heat-exchanger concept.

SILICA GEL

Assume the silica gel beds used in conjunction with the CO_2 control system can also be used for humidity control, then the weight of the chemical and its containers need not be applied as a penalty to the H_2O removal system.

Vacuum Regeneration

The water removed is lost to space during vacuum regeneration. Therefore its weight is:

1.87 lb/man/day x N x 3 men = 5.61 lb/day

where:

N = the number of days.

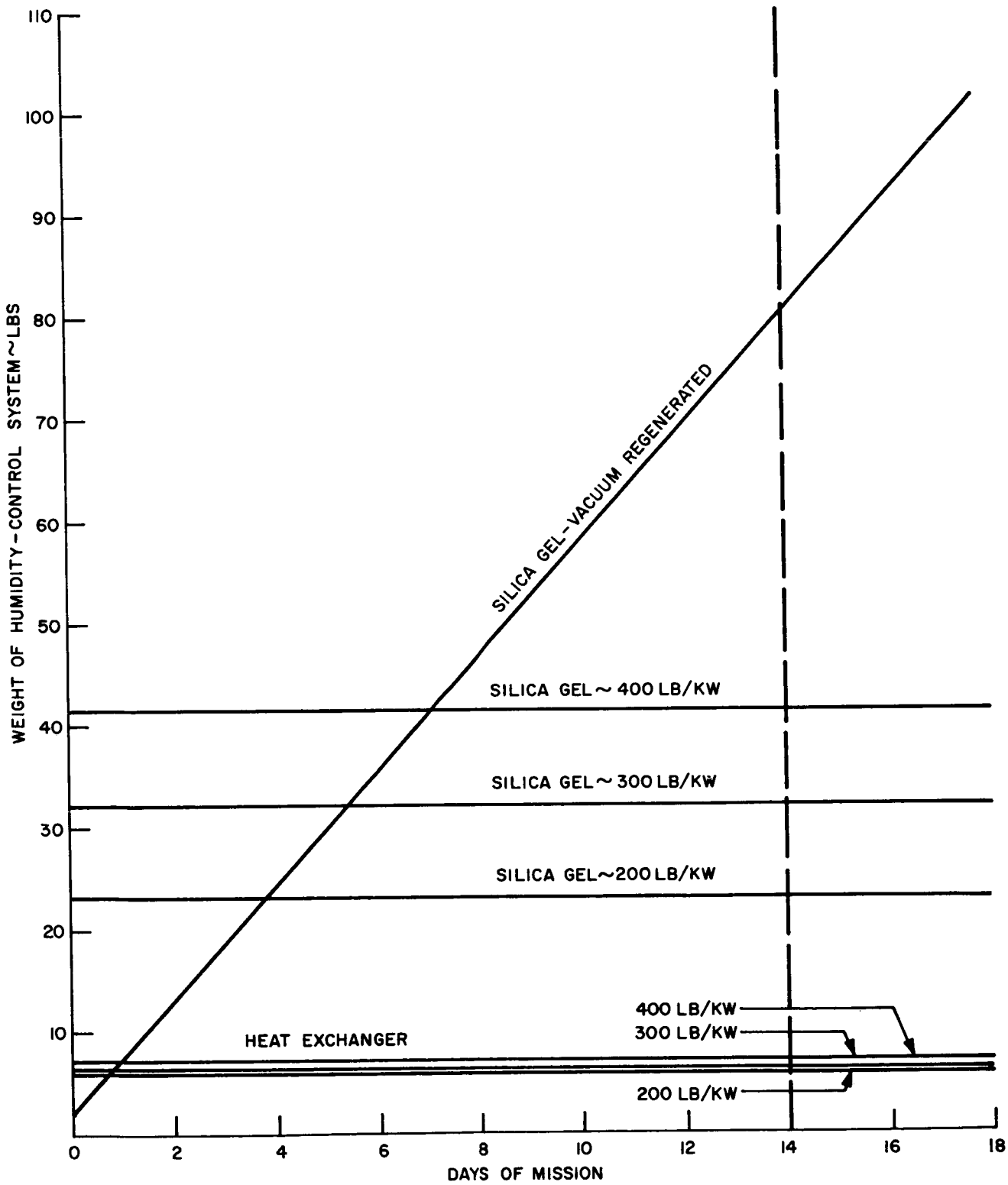


Figure XI-15-1. Humidity control systems for 3-man mission

The vacuum regeneration device is estimated to weigh two pounds.

$\therefore$ system weight = 2 + 5.61 N lbs.

Thermal Regeneration

Heat of regeneration for silica gel is 1400 Btu/lb H_2O recovered, or:

$$\frac{1400 \text{ Btu/lb } H_2O \times 1.87 \text{ lb. } H_2O\text{/man day} \times 3 \text{ men}}{24 \text{ hrs/day} \times 3.41 \text{ Btu/watt in}} = 96 \text{ watts}$$

Since the CO_2 system requires a thermal purge once every five cycles, four-fifths of the regeneration power can be charged to H_2O removal. Assuming 5 lbs and 5 watts for the collection and storage equipment, the system will weigh:

$5 + 4/5\ (96 + 5)\ Z = 5 + 82\ Z$

Lithium Chloride

This material will absorb water in an amount equal to 60 percent of its own weight. The water is not easily reclaimed, and its loaded weight must be assigned to the system. Adding a 10 percent weight for the container, the system weighs:

$1.1 \times 2.67\ (1.87 \times 3)\ N = 16.4\ N$

HEAT EXCHANGER

The thermal balance of the cabin is such that some heat must constantly be removed in the air-conditioner. Since, for the desired cabin conditions, moisture will condense on the cooling coils, no weight penalty is assigned for the cooling of the atmosphere to condense H_2O. Therefore, the only weight assigned to the water removal system is 5 lbs and 5 watts for collection and storage devices:

$5 + 5\ Z$ lbs.

Figure XI-15-2 shows that the expected range of coil surface temperatures is more than adequate to maintain the humidity within the desired range.

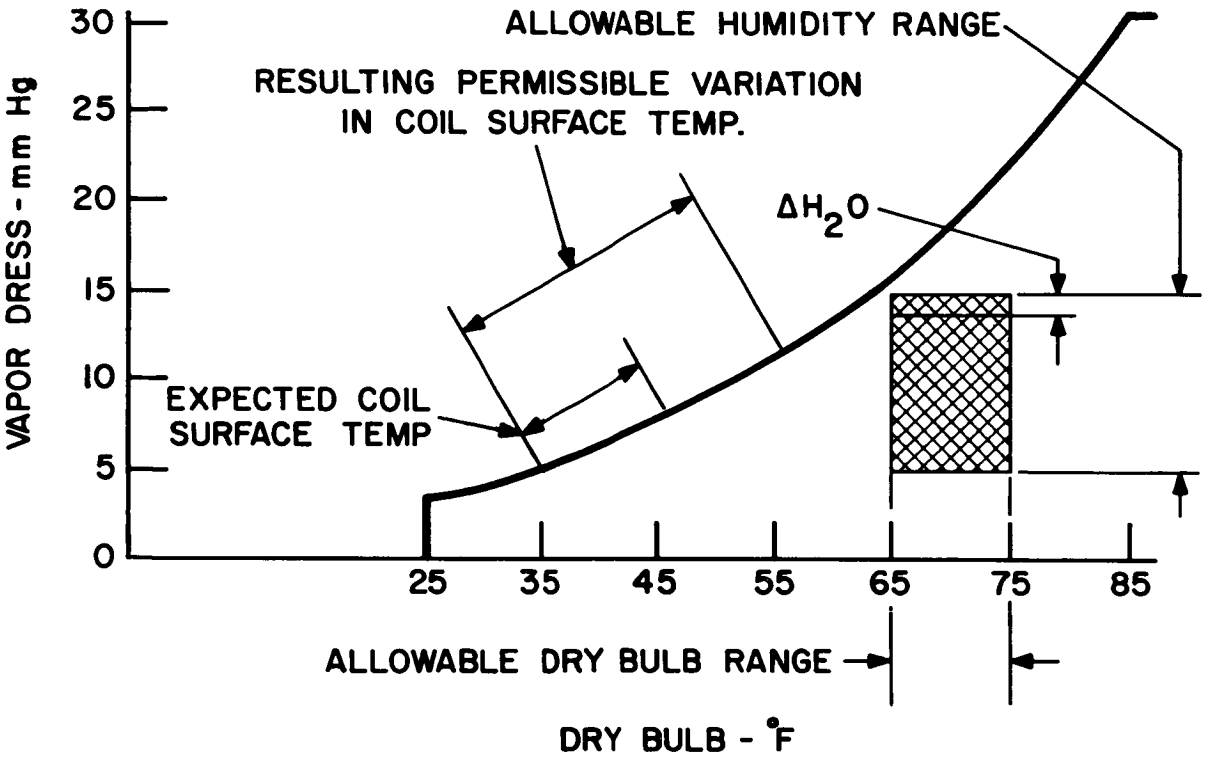


Figure XI-15-2. Expected range of coil surface temperatures

16. NOXIOUS AND TOXIC GAS CONTROL

The General Electric Company has initiated a program to determine what noxious and toxic gases will be added to the cabin atmosphere of space vehicles; their evolution rate; their M.A.C. for continuous exposure; and how to deal with them. Chapter IV includes some early information from this study. The likely sources of these gases are outgassing from materials and production by metabolism and equipment. The radiation environment of space will probably add to the problem by breaking down materials and gases, as well as causing combination of harmless gases to form noxious and toxic products.

A number of methods are available for removing those gases which cannot be prevented from being added to the cabin and which can accumulate to a harmful level. Catalytic combusion, using Hopcalite, effectively removes hydrogen, carbon monoxide, and certain hydrocarbons. Adsorption on activated charcoal is effective in removing a number of hydrocarbons, especially the odor-producing aromatics. Both of these are going to be included in the APOLLO conditioning system.

Although it is not anticipated that any noxious or toxic gas which cannot be controlled by the Hopcalite and activated charcoal, will accumulate to a dangerous level, enough gas reserve is included in the oxygen and diluent supply to permit purging of the cabin, should this occur.

Fortunately this area is one which can be completely explored during the system development phase of the program. As soon as the materials which will be contained in the vehicle are fairly well defined, a system test can be conducted. Laboratory-type instruments, such as gas chromatographs, can be used to determine precisely if the concentration of any gas — not accounted for, based on the results of previous study — builds up to an annoying level. If so, and if possible, the source of the offending gas will be removed. If removal is not possible, then a specific control mechanism can be added to the cabin environmental control system.

The mass spectrometer, which is to be used to sense the constituents of the cabin atmosphere during flight, has the inherent capability of detecting small concentrations of gases. Consequently, it can be used to obtain broad-spectra gas analysis data which can be telemetered to the Earth for analysis by a trained spectroscopist. This scheme is explained in more detail in Section XI-21.

17. PARTICULATE MATTER CONTROL

In any confined area, especially where human activities transpire, there is always an accumulation of impurities or foreign matter in the air. Clean air cannot by itself eliminate infection, but unclean air cannot be tolerated in a space cabin. Adequate cleaning of the air by an appropriate method is mandatory for a closed ecological system. Micro-organisms which on earth would be of no concern might, in the space cabin, become highly hazardous. Figure XI-17-1 shows various interesting characteristics of airborne particulate matter as a function of particle size. A point worthy of note is that the settling rate of 0.1 to 0.5 micron particles — the size range which is harmful — is so low that almost any air motion at all will keep the particulate matter airborne. Consequently, the loss of normal gravitational precipitation, due to the zero gravity environment of the APOLLO vehicle, should not significantly increase the hazard from airborne particulate matter.

PARTICULATE MATTER SOURCES

The mechanisms of particle production are primarily:

(1) Evaporation and recondensation

(2) Chemical dissociation

(3) Mechanical generation

(4) Mechanical dispersion and evaporation

(5) Conversion of gases.

Evaporation and Recondensation

Whenever the vapor pressure of a substance exceeds the partial pressure of that substance in its ambient, molecules escape from the material into the ambient. As the vapor pressure is a function of the temperature, if a volatile material is heated, molecules can escape. If the temperature of the ambient is lower, then the molecules will

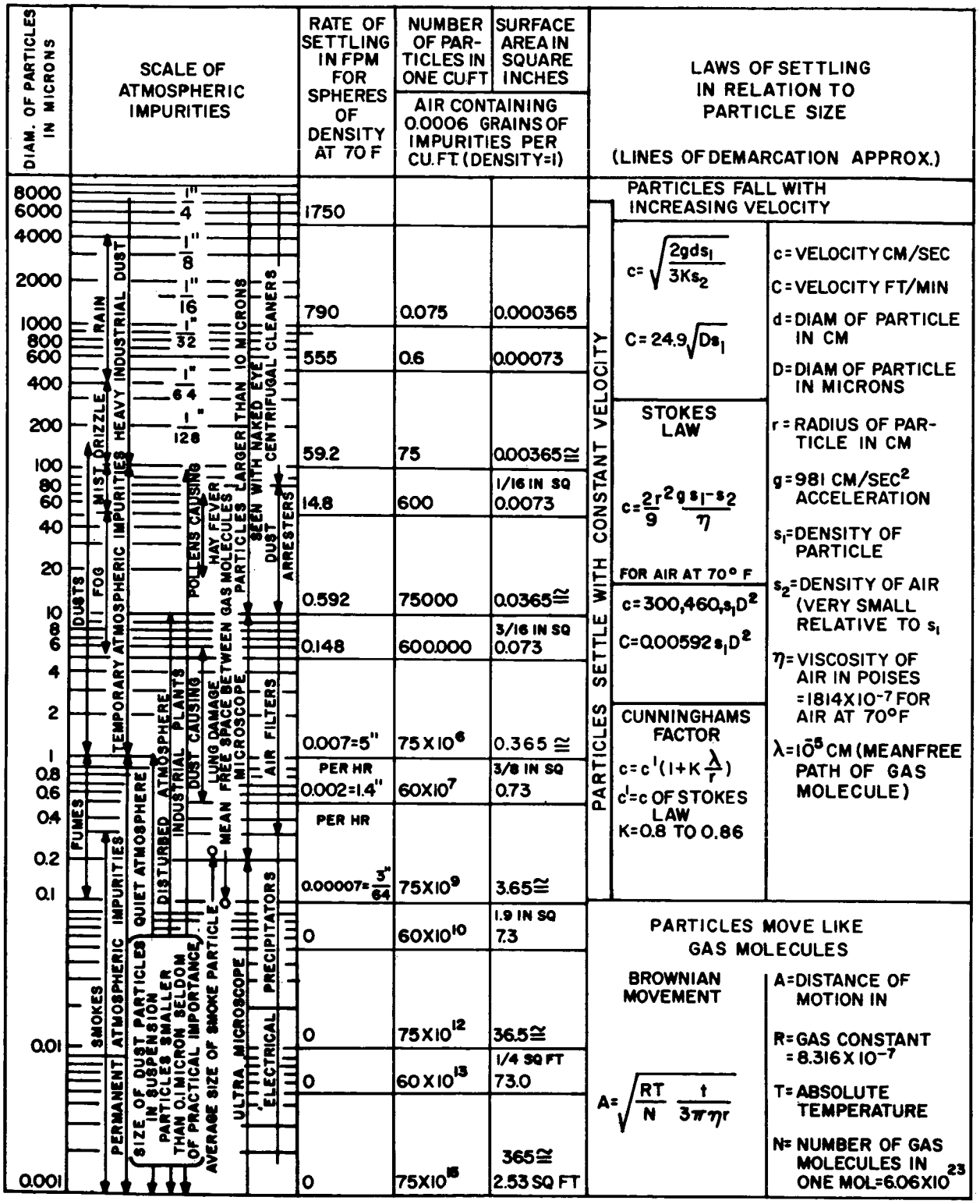


Figure XI-17-1. Size properties of airborne particulate matter

recombine and become airborne particulate matter. In general, particles so produced are quite small with few larger than 0.1 microns in diameter.

An example of this mechanism is the production of many airborne particles during an electrical discharge (e.g. motor brushes, relay contacts opening), when the electrode material is evaporated and recondenses into minute particles. Many particles are also formed by this process by the smoking of cigarettes.

Chemical Dissociation

The size and concentration of particles produced by this mechanism are extremely difficult to predict or control. Take, for example, the chemical dissociation of carbonyls. Carbonyls are gaseous compounds of carbon monoxide with metals. They can be detected in metallic CO_2 tanks which were sealed for long periods of time. It is suspected that they are also formed by combustion. When released into air at normal temperatures, the carbonyls react with the water vapor present to release metal atoms which form small metallic particles.

Mechanical Generation

This mechanism includes any action, either natural or man-made, by which particles are mechanically produced and distributed into the air, such as the rubbing and consequent abrasion of two surfaces (such as cloth on cloth, causing lint). The particles generated by this mechanism are physically torn from the parent material and dispersed into the air. Liquid sprays are also considered in this category. The particles so produced are generally sized from 0.1 to 100 microns.

In determining the ease of production, the mechanical properties of the material, such as crystal structure, density, and bonding forces, are important for the solids; viscosity and density are the more important factors for the liquids.

Mechanical Dispersion and Evaporation

This mechanism by which particles can be produced involves the removal or evaporation of material which bonds the particulate matter to a larger mass.

Conversion of Gases

The formation of particles from the chemical change of a gas brought about by oxidation has been demonstrated in the laboratory and in the free atmosphere. The importance of this mechanism has only recently been realized. By chemical change, gases in the air may form other compounds which tend to be solid or liquid under normal pressures and temperature. This process occurs in nature and in the smogs of Los Angeles. An excellent example of this type of production is the conversion of SO_2 gas into SO_3 by reaction with ozone and the subsequent formation of H_2SO_4 droplets when water vapor reacts with SO_3. By this process, very small droplets in the range from 10^{-7} to 5×10^{-7} cm in diameter are produced[(1)]

Aside from design problems, (such as providing lint-free clothing and preventing human excreta from contaminating the atmosphere) the main cause of generation of airborne particulate matter will be smoking and shaving of the face.

Cigarette smoking will cause a serious problem in the closed ecological system of the space cabin. One-quarter of a cigarette will produce around 140,000 particles/cc in a volume of 300 cu ft. The particles vary in size over a range of 0.1 to 0.01 micron in diameter[(1)].

Shaving, either by an electric shaver or the soap-and-razor method, will introduce particles into the cabin atmosphere. Since the size, density, or distribution of these hair and skin particles varies with man, an exact analysis of the particle production is difficult.

As previously mentioned the particle sizes which are most detrimental to men are those in the range of 0.1 to 0.5 microns in diameter. Consequently, particles in this size range must be removed with a relatively high efficiency. The filtering system will be designed so that there will not be excessive lung irritation, coughing due to airborne particulate matter, eye irritation due to airborne particulate matter, or an excessive accumulation of airborne bacterial spores and micro-ogranisms.

---

(1) F. W. Van Lurk, Airborne Particle Production.

Some possible method of filtering the air are:

(1) Centrifugation

(2) Particulate filtration

(3) Electrostatic filtration

(1) Centrifugation is used primarily to collect the coarser particles in the air. The fact that the precipitation rate, in a one g field, is so low for particles in the 0.1 to 0.5 micron range, is evidence that centrifugation is not particularly suitable. In fact, (see Figure XI-17-1) centrifugation is not ordinarily used for particles less than 80 microns in diameter.

(2) Particulate filtration is widely used and highly effective. Table XI-17-1 lists some significant characteristics of some commercially available filter materials. It can be seen that extremely high filtering efficiencies can be obtained for the small particle sizes.

(3) The electrostatic filtration process ionizes the dust particles so that they can be collected on high-potential plates. This will probably cause some ozone production, which if significant in quantity can be highly undesirable.

From the data presented, it appears evident that the use of particulate filters will do an adequate job of removing the airborne particulate matter which can be expected in the APOLLO vehicle. Not only will the filters remove with a high efficiency the particles in the 0.1 to 0.5 micron range but they will, of course, remove the larger particles with a still higher efficiency. Vomitus can cause a real problem with any filtering system. About the only way of handling this problem is to carry a spare filter so that the contaminated filter can be replaced.

Of the filter materials listed in Table XI-17-1, the Aerosolve 95 filter media has been selected for use in the APOLLO vehicle. It combines adequate filtering efficiency with relatively low pressure drop. The filter design contemplated would occupy a volume of one cubic foot, weigh about four pounds and cause a pressure drop of about 0.20 inches of water when filtering the 274 CFM planned for the APOLLO thermal control system. See Section XI-19.

TABLE XI-17-1. CHARACTERISTICS OF FIBROUS FILTER MATERIALS[2]

Code	Trade Name	Material	Diameter of Glass fiber μ	Efficiency %*	Rating Test	Rated Velocity, Ft per min.	Pressure Drop In H_2O !	Efficiency vs Bacterial Droplet Nuclei %	Efficiency vs Dustborne Bacteria %
H 93	CM 114	Glass-fiber paper	1.0	99.97	DOP ±	5.5	0.90	99.97	100
H 98 B	Micretain	Glass-asbestos paper	1.0	95	DOP	5.0	0.40	96.7-98.1	100
Ceramic	Fiberfrax	Aluminum silicate paper	2.4	83	DOP	5.0	0.18	97-98	99
50-FG	Filterdown	Glass-fiber mat	1.25	99-95	NBS #	21.0	0.48	87-99	99
C 95	Aerosolve 95	Glass-fiber mat	1.0-1.1	90-96	NBS	23.0	0.35	89-98	99
C 85	Aerosolve 85	Glass-fiber mat	0.8-0.9	80-85	NBS	25.0	0.22	54-69	

* During useful life of filter

± Dioctyl phthalante 0.3 μ smoke

\# National Bureau of Standards discoloration test

! At rated velocity

[2] Henry F. Allen, Air Hygiene for Hospitals

18. ION CONTROL

INTRODUCTION

A variety of results have been observed to be caused by ions in inhaled air, on the physiological and behavioral aspects of humans and animals. Experiments have revealed effects that have ranged from those having little significance to observations which were somewhat startling. The "small ions" of interest here, have diameters of approximately 10^{-7}cm and mobility of 2cm/sec/volts/cm. Reports indicate generally that negative ions have a normalizing effect on subjects under some stress with respect to activity, reaction time, fatigue, reliability of performance, vision, and emotional stability. Positive ions, on the other hand, seem to cause adverse effects. Information regarding ion effects has been compiled in a report by Allan H. Fray (GE R59ELC121). Some brief examples from this report are listed below:

1. Sensory sensitivity in vision and audition have reportedly been affected adversely by positive ions. There has been statistically significant correlation between changes in ionization and tinnitus (ringing in the ears).

2. Speicher (1959) found that positive ionization increased the reaction time of some people, whereas negative ionization reduced the reaction time of some people. Vytchikova and Minkh (1959) reported a decrease in the reaction time of athletes of approximately five percent as a result of daily 15-minute inhalations of negative ions.

3. There is evidence to indicate that air ions affect the biochemical processes involved in energy production. Nielson and Harper (1954) reported a significant reduction in the amount of succinoxidase activity in the rat adrenal gland. The succinic oxidase system is a chain of enzymes which catalyze the oxidation of succinic acid. This transformation is important in the terminal oxidation and interconversion of carbohydrates, fats, and amino acids. Since this is a key reaction in the Krebs cycle, which releases energy, it can affect activity.

4. Winsor and Beckett (1958) reported that five minutes of positive ionization reduces breathing capacity to 25 to 35 liters/minute. This is a reduction of approximately one-half in humans. Winsor and Beckett, however, had a funnel over the subject's face

and the procedure as described by them would indicate that considerable restrictions were placed upon breathing. Krueger and Smith's (1958) finding that positive ions cause a constriction of trachea would indicate some possible connection for the finding of Winsor and Beckett.

5. More extreme results in tests have demonstrated that, in the absence of ions and particles (pure air), pigeons, rats, guinea pigs, rabbits, and dogs had all died within 12-1/2 days - the majority within seven days. (Prof. L. A. Tchijevsky by T. A. Rich, G-E Report No. 60GL39).

This experiment, however, was contradicted by tests conducted under similar conditions by T. A. Rich at the G-E General Engineering Laboratory. Mice had been subjected to both deionized air and filtered air over a period of seven weeks and all lived in a state of apparent good health.

ION GENERATOR DESCRIPTION

A small radiation source, such as tritium, is a promising possibility for small ion generation because of its small size, high degree of ion-generation capability, and safety. Two other methods of ion generation are corona discharge and ultra-violet radiation. These methods are not considered because of the possibility of producing ozone, which is highly toxic. A tritium source which is approved by the AEC, was developed jointly by Wesix Heater Company in San Francisco and United States Radium Corporation. It has the following specifications:

Dimensions:	Diameter	- 2 inches
	Depth	- 1 inch
	Weight	- 4 ounces (source only)
Ion generation		- 1 billion ions/sec
Radiation		- 50 millicuries; zero rems
Life of source		- 12.5 years

Power Requirements: - The ion source constantly emits Beta Rays. However, in order to repel the negatively generated ions away from the source, its

electrode is held at negative potential. The following power requirements are estimated to create the electrostatic field.

Voltage	approximately 400 d-c, negative
Load	10^{-8} amp or less

Power Supplies: In order to generate the electrostatic field, the following power supplies are available:

Special Van Der Grinter batteries weighing about 6 oz total with a rating of 100 volts/inch

U. S. Radium also builds their own power pack which weighs about 4 pounds and operates from 115 volts, AC

OPERATION

The following general assumptions are made: Cabin volume is about 300 cubic feet. Human air intake is approximately 150 cc/sec. The generator source is placed about two feet away from the head.

A tritium source in a plastic mounting is attached to an insulated base. An electrode 1-3/4 inches in diameter is placed about 1/64-inch behind the source and is maintained at negative d-c potential. The tritium source emits soft Beta particles which produce ion pairs with a radius of about 5×10^{-8}cm and a mobility of 2cm/sec/volts/cm. The small negative ions traverse the electrostatic field in the direction of the astronaut who is at zero potential. Positive ions are attracted to the negative potential level. This is shown schematically in Figure XI-18-1. The average energy of the Beta particles is 5-1/2 kev and source is capable of generating approximately one billion ions a second. Emission distance of the source is approximately 2 millimeters in air.

OTHER VARIABLE CONSIDERATIONS

It is expected that the astronaut would be subjected to a concentration of ions in the vicinity of the head which is considerably less than at the source. Small ions disappear

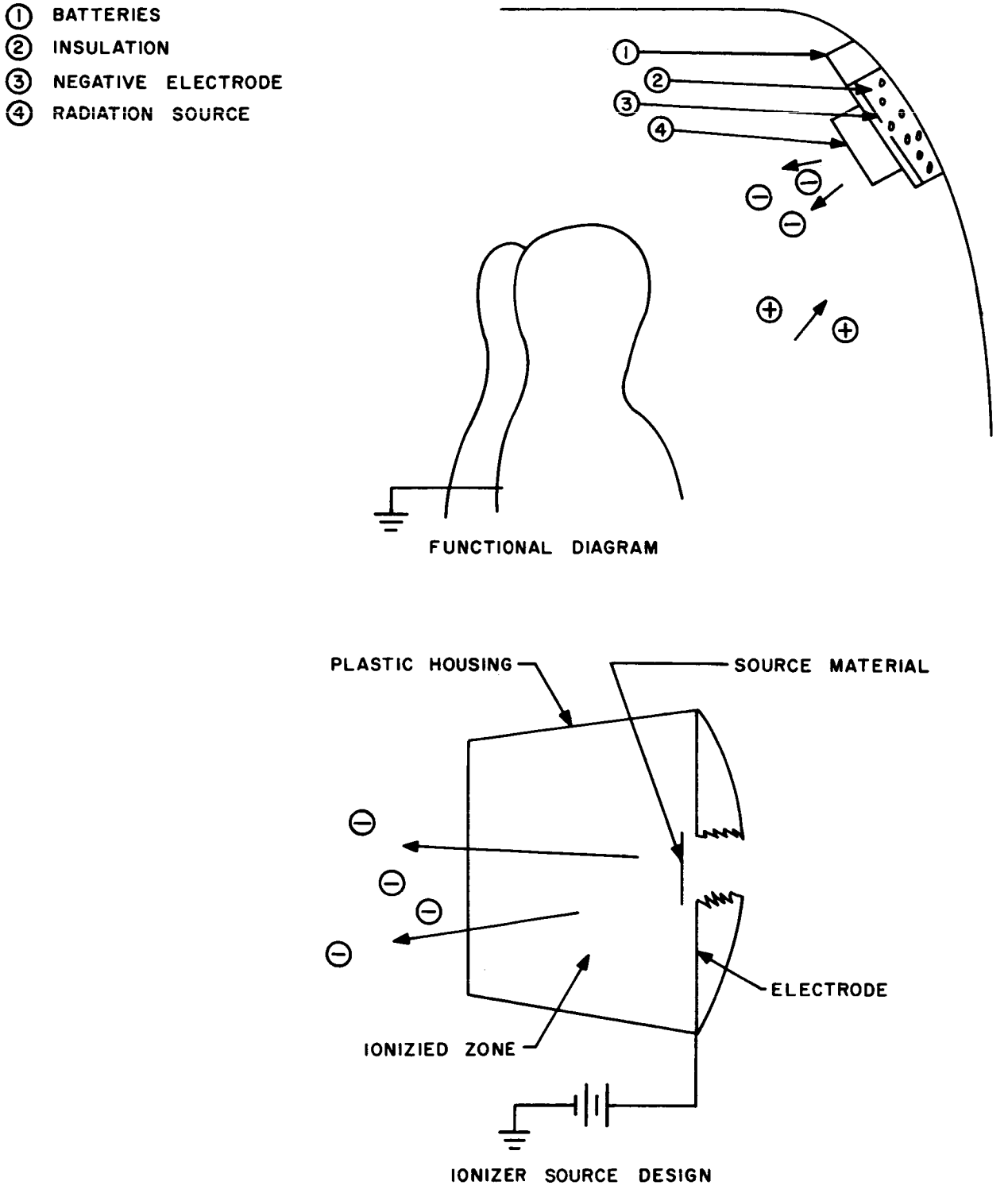


Figure XI-18-1. Ionizer source design and functional diagram

in transit because of positive and negative recombination, collision with larger particles and surface adsorption.

It is difficult to state with certainty what the ion concentration will be in the area of breathing influence. While it is relatively simple to generate ions, distribution problems in a cabin could be an extremely complex problem. For example, even with an increase in density of several thousand ions per cc at the source generator, little increase of ion density may occur in other parts of the same room because the ion concentration will be affected by other variables such as strary electrostatic fields produced by other equipments, purity of atmosphere, and air currents. In order to obtain approximate measurements of small ion concentration in the area of breathing influence, an ion-chamber mock-up would be required to simulate cabin conditions. An electrode of some potential would simulate the nose and mouth regions. Small ions would be injected into an air stream, which would approximate the capacity of human intake. Ions impinging on the electrode could then be amplified and measured by special sensitive measuring devices.

Another consideration is the question of whether continuous or pulsed ion generation is necessary. Some findings indicate that the effects of negative ions may be cumulative. If this is true, continuous ion generation would be obviated.

19. THERMAL CONTROL

CABIN THERMAL CONTROL

General

The cabin thermal control subsystem will be capable of maintaining the cabin at any desired effective temperature between 65 and 75 F. The effective temperature interrelates the thermal physiological effects of atmospheric dry bulb, p H_2O, gas velocity, and wall temperature. This temperature will be controlled by the cabin heat exchanger regulating the pH_2O through condensation and the dry bulb through sensible cooling.

The methods of controlling cabin temperature which have been studied for application in the APOLLO vehicle are:

1. Heat transport concept
2. Vapor compression concept
3. Expendable refrigerant and integrated cryogenic concepts

Each of these systems are discussed in this section and trade-offs made to select the system on the basis of weight, performance, power required, volume, integration possibilities, and problem areas.

The cabin cooling load consists of the metabolic heat output of three men plus an average of about 500 watts electrical heat for the mission duration. For the mid-point effective temperature, the heat input per man is taken as follows:

Sensible Heat	285 Btu/hr
Latent Heat	125 Btu/hr
TOTAL	410 Btu/hr

The heat load for the cabin is then:

$= 3 \times 410 + 500 \times 3.413$

$= 2940$ Btu/hr

The thermal environment external to the vehicle indicates a low average outer skin temperature which means that some of the cabin heat load will be removed by conduction and radiation through the vehicle skins and thence to space. Two factors restrict this passive transfer of the internally generated heat to space. The first is the amount of thermal insulation required between the cabin and the external surface of the vehicle to prevent an excessive cabin temperature rise during re-entry. The second is that a very low convective heat transfer coefficient exists, during weightless flight, between the cabin air and the cabin wall. Therefore, for the purpose of this analysis, the heat carried out through the wall is assumed to act as a cabin cooling system safety factor, and the heat exchanger is sized to carry the entire heat load. The balance of the approximately 3000 watts of internally generated power will be transferred directly to the cooling system without first being put into the cabin air. This will result in a weight and power saving in the cabin heat exchanger and a saving in radiator size, due to the fact that the heat will be radiated at a higher mean temperature—and the radiator size is an inverse function of the fourth power of the absolute temperature at which the heat is being radiated.

The volume of air to be circulated for conveying the heat to the cabin heat exchanger depends, for a given heat load, on the magnitude of the temperature drop. The lower limit is set by the 32 F limitation on the heat exchanger surface temperature to prevent freezing of the water condensed on the cooling surface. It is desirable, of course, to have as low an air flow as possible in order to limit the required fan power.

Discussion of Possible Thermal Control Concepts

LIQUID TRANSPORT CONCEPT

The liquid transport system derives its name from the fact that a suitable fluid is used as the means of conveying the heat to be rejected into space. The fluid is circulated by a pump through coils within the air conditioner and then directed through connecting tubes to the electrical cooling system and thence to the radiator on the outer skin of the vehicle, where heat is removed from it and radiated into space. The system is shown in the schematic diagram, Figure XI-11-1. The only additional component part required would be a thermostatic control device to maintain the temperature within the cabin at the desired level.

An advantage of the liquid transport system is that the various components may be located at different places within the vehicle without penalizing the efficiency of the system with other than the additional weight of the interconnecting tubing between components. It is possible to remove heat from several sources and convey it to a common radiator by use of a single pump. In addition, since this type of system does not require the heat transfer fluid to undergo any change of state, (as is the case, for instance, in a vapor compression cycle) it is not gravity dependent so far as its operation is concerned. The primary disadvantage of the liquid transport system is that it can reject heat only when the external sink temperature is lower than the temperature of the cabin temperature. A low-temperature sink is available in outer space, however, and, by proper orientation of the radiator during a mission, it is possible to take advantage of this low-temperature sink. It is intended to orient the APOLLO vehicle so that the radiator will be in the shadow for practically the entire mission.

To perform the cabin air conditioning, calculations indicate the following system parameters to be required:

Heat rejection	2940 Btu/hr
Air circulation	274 cfm
Fan power required	160 watts (two 80-watt blowers)
Size (heat exchanger)	0.25 cu ft
Weight (less radiator)	25 lbs

Ground Cooling

Providing cooling for the vehicle on the ground, when required for testing or pre-launch activity, can be readily accomplished by chilling the external radiator surface from a ground support cooling system (such as a vapor compression cooler or a portable water chiller unit) or by connecting a portable chiller unit directly into the fluid lines, thereby by-passing the radiator completely.

Powered Flight Heating

The transient heat loads resulting from launch will require special consideration. The liquid transport system, as well as all systems in which the heat load to be

rejected is radiated to space, will not operate until the radiator has cooled, following powered flight. It is expected that the launch heat load can be absorbed in the mass of the vehicle, and prechilling of the vehicle (to about 60 to 65F) prior to launch may be necessary.

Advantages

The liquid transport method of heat removal will not require the learning or developing of new technologies or hardware. It does not require large power inputs to operate compressors nor does it have the inherent problem of phase separation which exists in a two-phase vapor compression cooling system. The components will be similar or identical to commercially available parts which have proven the best combination of performance, weight, and reliability. In addition, the liquid transport method of removing cabin heat makes possible many alternates in the design; the optimization of which depends upon the total systems integration.

VAPOR COMPRESSION CYCLE CONCEPT

General Description of the System

This type of system is the one most commonly utilized in commercial and military non-flight applications. It derives its name from the fact that the working substance is carried from the low-temperature region to the high-temperature region by compressing it.

Figure XI-19-1 is a flow diagram for a vapor-compression air-conditioning system. The refrigerant, which for this application is Freon 12, passes through a circuit which is divided into high-pressure and low-pressure portions. The pressure is maintained at different levels in the two parts of the system by the expansion valve at one point and by the compressor at another. The function of the expansion valve is to allow the liquid refrigerant under high pressure to pass at a controlled rate into the low-pressure part of the system. Some of the liquid evaporates the instant that it passes the expansion valve, but the greater portion is vaporized in the evaporator at the low pressure which is maintained by the exhausting action of the compressor. In evaporating, each pound of liquid absorbs its latent heat of vaporization, the

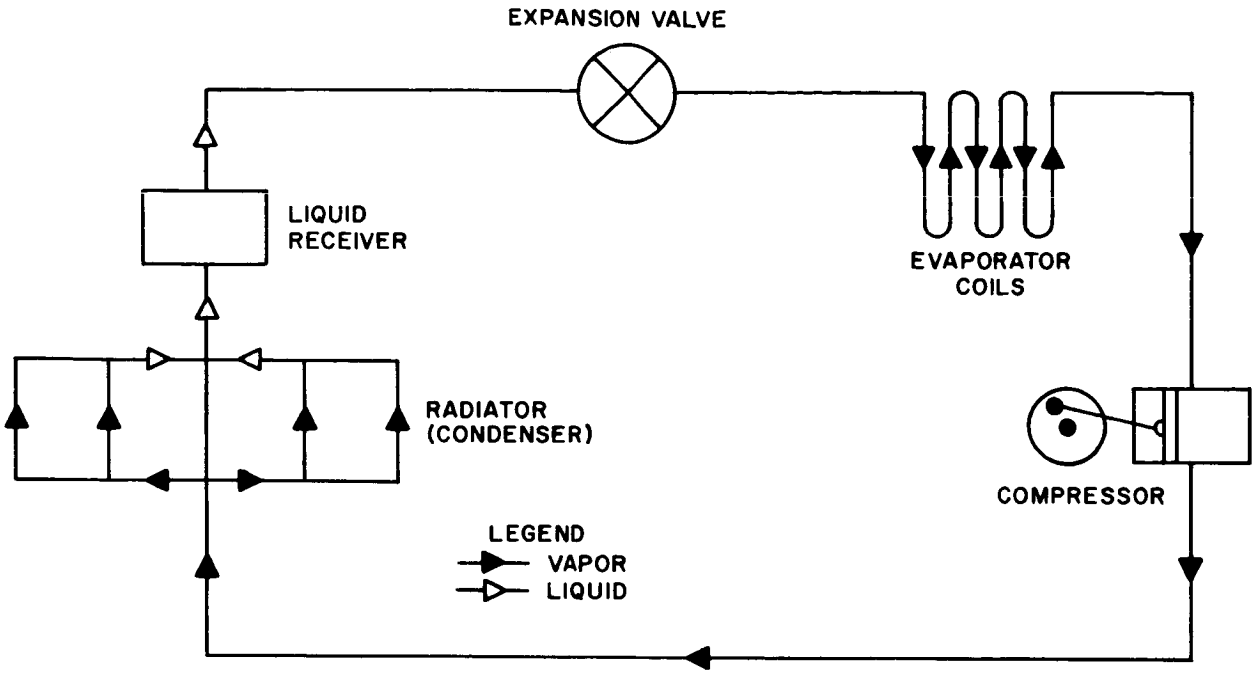


Figure XI-19-1. Vapor compression cycle, functional schematic diagram

greater part of which is conducted to it through the evaporator tubes from the air. Capillary tubing is now commonly used in the air conditioning and refrigerating industry to replace the more complex expansion valve.

The function of the compressor is to increase the pressure of the refrigerant vapor and discharge it into the radiator on the outside of the vehicle. The highly compressed, super-heated refrigerant vapor is then condensed and sub-cooled in the radiator. Because this vapor is at a high pressure it condenses at a comparatively high temperature and gives up the heat which is absorbed in the evaporator at the lower temperature, plus the heat equivalent of the work done on it by the compressor. In a space vehicle the heat to be rejected would be transferred by convection to the radiator tubes, from which it would be radiated to space.

The vapor compression system is similar to the liquid heat transport system, already described, in that the heat is ultimately rejected by radiation to space. The advantage of the vapor compression system is that the heat can be rejected at a higher temperature.

A system such as described above could be expected to have a coefficient of performance somewhere between two and four, depending on the temperature at which the heat is rejected from the radiator. The coefficient of performance (COP) is defined as the heat to be removed divided by the power required to remove the heat.

The vapor compression system necessary to perform the cabin air conditioning will require the following parameters at an assumed COP of three:

Heat rejection	3920	Btu/hr
Air circulation	274	cfm
Fan power required	160	watts (two 80-watt blowers)
Compressor power req'd	290	watts
Size (heat exchanger)	0.25	cu ft
Weight (less radiator)	50	lbs

An analysis indicates that the saving in radiator weight due to the fact that the heat is rejected at a higher temperature does not compensate for the power required to drive the compressor.

As mentioned previously, the use of the two-phase vapor compression concept entails the separation of the two phases prior to compression and prior to expansion.

The only advantage of a vapor compression system occurs where the orbit conditions are such that radiator orientation cannot be controlled, or a minimum of area is available for radiator surface. In this case, the vapor compression system, because of the higher temperatures at which it rejects heat, might be advantageous.

EXPENDABLE REFRIGERANTS AND INTEGRATED CRYOGENIC CONCEPTS

Expendable refrigerants are suitable primarily for very short missions and are not applicable for the APOLLO vehicle. If water were used as an expendable, the volume required to absorb both the electrical and metabolic heat of the APOLLO mission would be 3840 pounds. An evaluation has been made of an integrated cryogenic system which utilizes cryogenic hydrogen as a heat sink, then burns the hydrogen with sufficient O_2 to raise the temperature to the desired value, expands the products of combustion through a turbine to obtain electrical power, and uses the discharge for

attitude control. Besides the formidable control problem envisioned for this system, the weight required is prohibitively high. The system weights and size are given in Table XI-19-1.

Selection

The systems which have been investigated are summarized in Figure XI-19-2. As indicated here, the lightest and most reliable system for rejecting the waste cabin heat of APOLLO is the liquid heat transport concept.

Component Discussion

AIR CONDITIONER

The major components of an air conditioner system utilizing the liquid transport concept of cooling are:

a. Heat exchanger

b. Circulating pump

c. Space radiator

d. Expansion tank

e. Air blowers

f. Air distributor

g. Water collector

h. Controls

The coolant fluid is circulated through a conventional finned-tube heat exchanger by a small, low-power circulating pump and then directed to an external space radiator where the waste heat is rejected. An expansion tank is provided for thermal expansion relief of the fluid. Coolant temperature would be maintained at 30 to 35 F in keeping with good commercial practices. This coolant temperature would cause moisture to be condensed out of the air onto the coils and, in a zero gravity field, this moisture would migrate downstream with the air. A moisture collector would be provided at the outlet of the air conditioner. For ground testing where the moisture

TABLE XI-19-1. CRYOGENIC COOLING SYSTEM SUMMARY

Gas	Use	Wt. (lbs.)	Tank Wt. (lbs)	Vol. (ft^3)	Equiv Sphere Size (ft)	Total System Wt. (lbs)
H_2						5475 H_2 and Tank
	Power gen.	1340	3575*	1020	12.5	895 O_2 and Tank
	Extra for cooling	560 (455 at +200 max. temp.)				50 Prime Mover
						25 Valves & Contr.
						6445 Sub Total
O_2						380 Environ Control
	Power gen.	600	295*	22.4	3.5	8 Elect H/E
	Environ.	112	55*	4.2	2.0	- Sp Rad
						6833 TOTAL

* Filament wound epoxy tank.

CONFIDENTIAL

CONFIDENTIAL

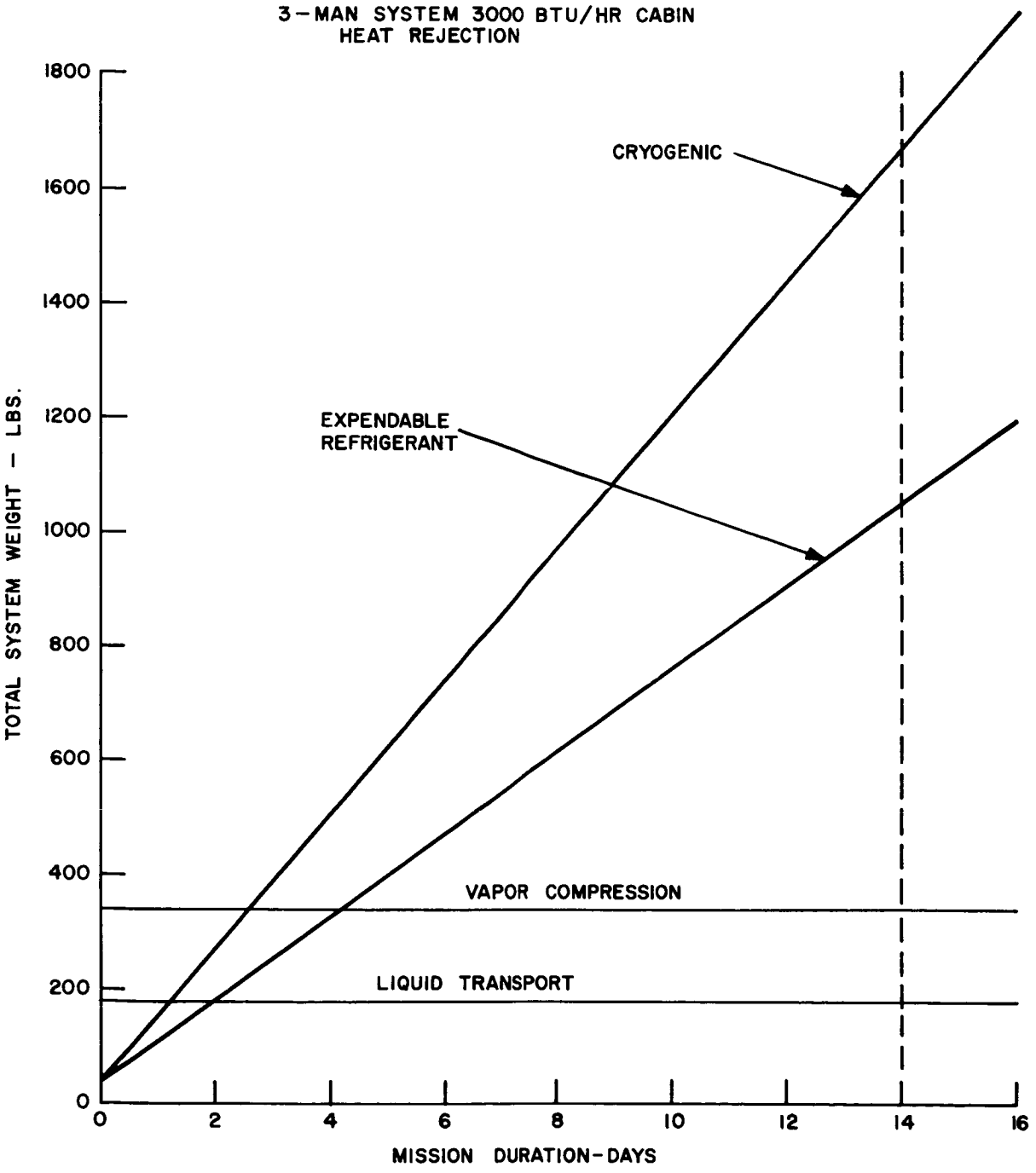


Figure XI-19-2. Cabin air-conditioning system weight comparison

would be influenced by gravity, the system could, if necessary, be oriented to make the air flow and gravity act in the same direction.

Heat Exchanger

The heat exchanger is an 8-inch by 12-inch by two-row, spined fin-and-tube heat exchanger. This is an unusually high performance surface developed by General Electric's Room Air Conditioner Department. The high performance is derived from the unusual amount of turbulence created in the air passing through the spined fins.

Air Conditioning Circulating Pump

The pump to be used for the circulation of the liquid transport fluid will deliver 2.04 gallons of fluid per minute. It is an integral pump-motor combination and operates on 28 volts DC. The pump is of the self-priming centrifugal type utilizing a neoprene rubber impeller. It weighs approximately 2.5 pounds and is 14 inches in diameter and 8 inches long.

Space Radiator

The calculations for the radiator indicate that it will be made in a number of parallel sections, one of which is redundant, and will weigh a total of about 27 pounds* and occupy an area of less than 200 sq ft.

Expansion Tank

The expansion tank, or accumulator, is a device which absorbs the changes in coolant fluid volume caused by temperature variations, preventing excessive pressure build-up which could possibly damage some portion of the circuit. The expansion tank also serves as a reservoir for extra coolant to make up fluid lost in the system through leakage. The design considered for the APOLLO vehicle is a closed cylindrical container which has a spring loaded piston with a sliding diaphragm, "bellofram" seal.

*This does not include low drag fairing skin weight, of which the radiator is a part, and would be required anyway, but does include heat transport fluid weight.

Air Blowers

The air circulation blowers will deliver 274 CFM through the air conditioner to the cabin and will be matched to the system for proper flow and pressure control. The blowers have been chosen on a basis of power, performance weight, size and reliability. Either, blowing alone, is capable of maintaining the effective temperature near the upper end of the permissible temperature range. It is expected that each blower will draw about 80 watts of 28 Vdc power.

Air Distributor

The delivery of the air to the cabin areas will receive close attention to provide the most uniform distribution possible and eliminate high velocity zones. A design which has been developed would deliver the air into a plenum type distributor section located at the top of the cabin. The duct can be acoustically treated to keep any air or blower noise which might reach the cabin below a minimum background if test results indicate that such acoustical treatment is required. In addition to the advantage of excellent air distribution, this cabin plenum results in a low airflow pressure drop and, consequently, the lowest possible fan power requirements.

The air leaving the plenum at the top of the cabin will flow down and be collected from beneath the foot area. All loose articles will therefore migrate to the foot area of the cabin. In effect, an artificial gravity, although weak, would be created. If necessary, portions of the air can be ducted off to provide local cooling to more critical hot spots.

Water Collection

The various methods of collecting moisture from the air are discussed in another section. The concept of utilizing the cabin heat exchanger for the control of the cabin humidity was established as best.

Several methods of catching and removing the water droplets from the air stream have been evaluated for this application.

(1) Passing all air through a porous adsorbent sponge and periodically squeezing to reclaim the water.

(2) Centrifuging in a reverse air flow bend and collecting on a pad, and

a. periodically squeezing the pad

b. "wicking" the water out into an accumulator

c. periodically circulating the pad out of the air stream for squeezing

(3) Collecting on a pad and centrifuging the pad for collection.

The system tentatively selected includes a belt-like pad, which is passed through the air stream to absorb the moisture thrown at it, due to the centrifuging action of the reverse air flow bend. The belt then passes through a roller squeezer where the water is collected and pumped to an accumulator. Circulation of the pad is intermittent so that the pad entering the squeezer rolls is kept as wet as possible and the volume flow to the pump is kept as high as possible.

20. FIRE DETECTION AND CONTROL

PHILOSOPHY

The APOLLO fire detection and control philosophy is to:

(1) Provide a system to detect imminent fires, before actual burning starts.

(2) Eliminate, insofar as possible, flammable materials from the cabin as well as those that produce objectionable gasses when heated. The proverb, "An ounce of prevention----" is very pertinent.

(3) Reduce fire hazard by keeping the atmosphere oxygen-partial pressure low and the diluent partial pressure as high as possible. See Figure XI-10-2.

(4) Monitor the design and installation of cabin equipment. Chart the possible causes of fire and provide protection for them.

(5) Provide both a fire-fighting procedure and extinguishers so that the fire can be quickly extinguished, should one start in spite of all precautions.

DETECTION

The products of a fire are heat, radiant energy, and products of combustion. Of these, both heat and radiant energy can possibly be used to detect the imminence of fire.

The usual temperature range of applicability of detectors sensitive to radiant energy is above 1000 F. Those that operate at lower temperatures have inordinate response times, since thermal radiation at low temperatures is small and consists largely of infra-red radiation. The usual flame temperatures are above 1000 F; therefore, detectors utilizing radiation will not satisfactorily indicate overheat conditions.

Since the indication of an incipient fire is necessary and since the detector must be operable at a set-point temperature below flame temperatures, the combustion outputs

of radiant energy and combustion products cannot be utilized. Hence, heat-sensitive elements appear to be the best method of detection.

Thermometric

Heat-sensitive detectors of this type utilize the phenomena of expansion of materials when heat is added. This expansion makes or breaks an electrical circuit and initiates an electrical signal to be utilized for indication and control. The material used may be liquid, gaseous, or bimetallic. Although systems of this type can be designed adequately, they have the inherent disadvantage of sensing the temperature at a point. For this reason, their use becomes prohibitive in terms of weight since many of them must be used to adequately cover the hazard areas expected.

Thermoelectric

Heat-sensitive detectors of this type operate on the principle of the Peltier effect. This effect is the production of a current flow when a thermal difference is applied at the junctions of a circuit of two dissimilar metals. A fire-detection system utilizing this effect suffers from the same disadvantage as the previously mentioned thermometric systems in that it will sense temperature only at a point. In addition, the output signal is in millivolts and requires amplification to operate an indicator.

Thermistors

Thermistors operate from the change of electrical resistance of a substance with temperature. This substance may be metallic or a semi-conductor material. The familiar resistance thermometer utilizes a metallic resistor. The resistance-temperature characteristic for metals is an increasing function. For small temperature changes, this function is linear and the rate of change of resistance is small. Ceramics have been developed which have a decreasing resistance temperature characteristic and whose rate of change with temperature is relatively large. The major disadvantage of this type sensor is contamination of the resistance element by oxidation. This necessitates enclosure of the element in a thermal well.

Detection System Description

Of the three types of heat sensing elements considered, it has been decided that thermistor-type sensors will be utilized for indication and detection of actual fire and incipience of fire.

The sensing element is one whose resistance is a decreasing function of temperature. The sensing element, upon detecting changes in temperature and rate of temperature rise, will initiate a signal to the control unit. The control unit monitors the resistance of the sensing element using a Wheatstone bridge measuring circuit in which the sensing element is one arm of the bridge, and a trip-setting adjustment resistor another. The bridge null detector is the first stage of a thermistor flip-flop circuit. As the element resistance decreases to the null-producing signal, the first stage of the flip-flop cuts off, causing the second stage to conduct. The second-stage collector current operates a relay whose contacts are used to operate the warning device.

CONTROL

Two methods of extinguishing fires are (1) spray an extinguishing agent onto the conflagration and (2) remove the oxygen atmosphere from the area of conflagration. The latter cannot be employed when the combustible material is self-oxidized.

Extinguishing Agents

One of the ways in which an agent acts is to cool the burning material by absorbing the heat. Another way excludes oxygen from the combustible material and thus smothers the fire. A third method is to enter into the combustion reaction chemically and hinder or halt its progress. Ideally, the agent should be capable of efficiently extinguishing any of the types of fire which is likely to occur. In addition to this, the agent must be non-toxic and should have no adverse effect upon the atmosphere regeneration system, either in its original or in its pyrolized state. The agent should also be as non-corrosive as possible in order to avoid damage to the cabin interior and components.

Table XI-20-1 illustrates the relative effectiveness of some possible vaporizing-liquid extinguishing agents. This effectiveness was determined by the minimum amount of agent required to extinguish a gasoline fire. Freon 13B1 and CO_2 are decidedly more effective than the others.

TABLE XI-20-1. RELATIVE EFFECTIVENESS IN FIRE TESTS

Agent	Formula	Effectiveness Relative to Freon 13B1 on a Weight Basis
Freon 13B1	CB_rF_3	100
Carbon dioxide	CO_2	85
Freon 12B2	$CB_{r2}F_2$	67
Bromochloromethane	CH_2B_rCl	45
Carbon tetrachloride	CCl_4	34
Methyl bromide	CH_3B_r	31

Table XI-20-2 compares the toxicity of these agents, as determined by the Underwriters' Laboratories. From these data, Freon 13B1, the least toxic, appears suitable. However, this is not the complete story. The data of Tables XI-20-1 and XI-20-2 are for unpyrolized vapor. The toxicity of these agents was also studied at the Army Chemical Center. In these tests, rats were exposed to air containing various concentrations of agents. In one series, the undecomposed vapor was used and in another the decomposed vapors, obtained after heating the compound, were used. Table XI-20-3 is a summary of results, indicating that Freon 13B1 gives off toxic vapors when pyrolysis takes place. These data indicate that carbon dioxide is the only suitable extinguishing agent of those listed.

A recently developed high-density foam (ABC Dry Chemical) which does not utilize water as a foaming agent has a distinct advantage in that it will remain in the area of the fire, thereby not contaminating the vehicle atmosphere. However, it utilizes an acid base and is highly corrosive.

TABLE XI-20-2. TOXICITY COMPARISON

Group	Definition	Examples
1	Gases or vapors which in concentrations of the order of 1/2 to 1 per cent for durations of exposure of the order of five minutes are lethal or produce serious injury.	Sulfur dioxide
2	Gases or vapors which in concentrations of the order of 1/2 hour are lethal or produce serious injury.	Ammonia, methyl bromide
3	Gases or vapors which in concentrations of the order of 2 to 2-1/2 per cent for durations of exposure of the order of one hour are lethal or produce serious injury.	Bromochloromethane, carbon tetrachloride, chloroform, methyl formate
4	Gases or vapors which in concentrations of the order of 2 to 2-1/2 per cent for durations of exposure of the order of two hours are lethal or produce serious injury.	"Freon 12B2", dichlorethylene, methyl chloride, ethyl bromide
Between 4 & 5	Appear to classify as somewhat less toxic than Group 4.	Methylene chloride, ethyl chloride
5a	Gasses or vapors much less toxic than Group 4 but more toxic than Group 6.	"Freon-11", "Freon-22", carbon dioxide
5b	Gases or vapors which available data indicate would classify as either Group 5a or 6.	Ethane, propane, butane
6	Gases or vapors which in concentrations up to at least about 20 per cent by volume for durations of exposure of the order of two hours do not appear to produce injury.	"Freon-13B1", "Freon-12", "Freon-114"

TABLE XI-20-3. APPROXIMATE LETHAL CONCENTRATION FOR 15-MINUTE EXPOSURE, PPH BY VOLUME

Agent	Formula	Not Heated	Heated
Freon 13B1	$CBrF_3$	800,000	14,000
Carbon dioxide	CO_2	658,000	658,000
Bromochloromethane	CH_2BrCl	65,000	4,000
Freon 12B2	CBr_2F_2	54,000	1,850
Carbon tetrachloride	CCl_4	28,000	300
Methyl bromide	CH_3Br	5,900	9,600

An inert gas, such as nitrogen, may be used as the extinguishing agent and would be safer even than carbon dioxide insofar as the occupants are concerned. However, it would be less efficient.

Oxygen Removal

An effective method of combating fires on a space vehicle is to dump into space the atmosphere from the compartment in which fire exists, thus removing all oxygen from the area around the fire. This procedure requires that the atmosphere be replaced. It also requires that the astronaut, after having been protected against the vacuum in the cabin, be mobile in order to remove the cause of the fire. Because the astronaut must protect himself before dumping the cabin atmosphere, the suitability of this scheme is dependent upon how quickly he can protect himself with the secondary pressurization system.

It is interesting to note that, in a zero-g field where natural convection is absent, the fire will tend to be self extinguishing. The oxygen molecules required to support combustion will have to diffuse through the inert (from a combustion support standpoint) gas cap surrounding the fire, composed of the diluent gas and the products of combustion. Consequently, oxygen removal will, to some extent, be provided by simply shutting off all equipment causing forced convection.

Selection of Control Procedures and Equipment

The fire-control procedure contemplated for the APOLLO vehicle will be to shut down all non-essential electrical equipment, including all blowers upon a signal from the fire detection system. This will automatically stop the removal of carbon dioxide from, and the addition of oxygen to, the cabin atmosphere. This can also be accomplished manually by striking a single switch.

A cooling and fire extinguishing foam, consisting of carbon dioxide and some physiologically inert and non-conducting viscous liquid, will then be sprayed from the fire extinguisher (one of which is located in each module) on the fire. The foam will leave the nozzle of the extinguisher at a relatively low velocity, so that it will adhere to the burning or smoldering material.

21. SENSING AND CONTROL SIGNAL GENERATION FOR CABIN ATMOSPHERE

INTRODUCTION

Two techniques for performing the title function are being considered with regard to performance, weight, volume, power requirements, reliability, and "fail-safe" indication characteristics. A further consideration will be the availability of task time by the crew for performing calibration functions. Of the two techniques, one involves the use of the mass spectrometer as an omni-gas sensor; the other makes use of infrared and magnetic susceptibility techniques for monitoring CO_2 and oxygen partial pressure respectively. Each system contains back-up sensing for vital gases in the event the primary system fails. Of course, if weight, volume and power requirements permit, reliability may be achieved by redundancy in either system. The primary system which makes use of the mass spectrometer appears to be the preferred system for this program.

Some of the toxic and noxious gases expected to accumulate in a man-inhabited sealed enclosure can be detected and their concentration measured with the mass spectrometer. One such gas is the hydrocarbon, methane. However, there may be toxic gases which will accumulate to significant concentrations, where the maximum allowable concentration is beyond the detection ability of the mass spectrometer. Studies running concurrently with the APOLLO design study and, subsequently, with the APOLLO vehicle design, will be expected to reveal these problem gases and their accumulation rates. Judicious selection of cabin materials, as well as the inclusion of adequate absorbance, can control many of these and remove them as problem areas. In the event certain of these single gases, vapors, or fumes remain a problem and if the primary atmosphere monitoring system is incapable of adequately measuring these concentrations due to insufficient sensitivity or resolution, additional specialized analytical instrumentation will be included in the on-board inventory.

MASS SPECTROMETER TECHNIQUE

A system making use of the mass spectrometer is preferred for this program. The specific instrument of choice is the time-of-flight instrument manufactured by Bendix. This instrument, because of its versatility in being able to analyze a gas mixture for all significant normal gases, and because of its relative independence of drift once careful calibration has established the normalization factors for each gas for the particular instrument, would certainly appear to be the instrument of choice as the sensing element in an automatic system such as would be desirable in the APOLLO vehicle. Briefly, the mass spectrometer determines the individual gases in a mixture by their mass-to-charge ratio. The concentration of a particular gas is determined by the relative number of ions generated.

A time-of-flight mass spectrometer designed to be borne aloft by space vehicles has been developed and built by the Bendix Aviation Corporation Research Laboratories Division at Detroit, Michigan. This instrument is an extremely compact and lightweight unit capable of delivering the complete mass spectrum of a gas mixture ten thousand times per second. This feature, of course, makes possible essentially simultaneous data on all gases in a mixture continuously.

The required flow rate of the gas sample through this instrument is only 2.63×10^{-4} cc per second. The instrument weighs approximately 12 pounds. It may be constructed in the shape of an "L", one leg of the "L" being approximately 19 inches and the other leg 11 inches. The electronic equipment for generating the ionization and gating pulses are contained in three transistorized modules, each module measuring 6 x 6 x 1-1/2 inches and the three units weighing about 3 pounds total. Of course, no vacuum pumps will be necessary, since the instrument can be exhausted by the vacuum of outer space. Approximately 17 watts at 28 volts is required to power the instrument. Some additional power will be required for the electronic devices, which are necessary to perform in-flight data reduction for purposes of generating control signals.

While no reliability studies on this instrument have been made, Bendix Aviation experience indicates a failure rate of not more than 2 or 3 out of 100 instruments once a

shake-down period has been completed. Carrying two instruments aboard the vehicle for driving the sensing and control system would thus achieve very high reliability.

The block diagram, Figure XI-21-1, indicates how the mass spectrometer would be used in a system for automatically maintaining the balance of the normal significant atmospheric constituents. One primary function would be to supply the signal utilized for oxygen partial pressure regulation. The mass spectrometer samples the air being exhausted from the space cabin prior to conditioning. The output of the instrument, consisting of voltages whose amplitudes are proportional to the individual gas concentrations, is applied to radar-type pulse gating circuits. The gate outputs are voltage pulses which program the circuits open in the proper time phase to register with the particular voltage peak in the mass spectrum. The integrated output from these gating circuits thus consists of a d-c voltage whose amplitude is proportional to the concentration of a particular gas. Ratioing this signal with the summation output of the mass spectrometer for all gases and multiplying by the signal from the sensor for measuring total pressure results in a continuous measurement of the partial pressure for each gas so selected. These signals can then be used for control, display, telemetry, and recording. Of course, for control purposes it will probably be necessary to effect impedance transformations in these output signals.

As an alternate to two mass spectrometers, back-up instrumentation for cabin atmosphere analysis employing polarographic techniques can be used. Polarographic sensors for measuring the partial pressure of CO_2 and O_2 would be available for emergency use sealed in airtight containers to preserve their shelf and operating life. For reliability, several of these cells would be carried in similarly sealed containers and broken open as necessary. The oxygen polarographic sensor can be calibrated by making use of the diluent gas supply as a zero calibration gas and the pure oxygen gas supply as a span gas calibrant. A siphon bottle type CO_2 cartridge can be carried for calibrating the CO_2 sensor. Subtracting the sum of the partial pressures of oxygen and CO_2 derived from these polarographic cells from the total pressure given by the cabin pressure sensor will result in a measure of the partial pressure of the diluent gas present.

CONFIDENTIAL

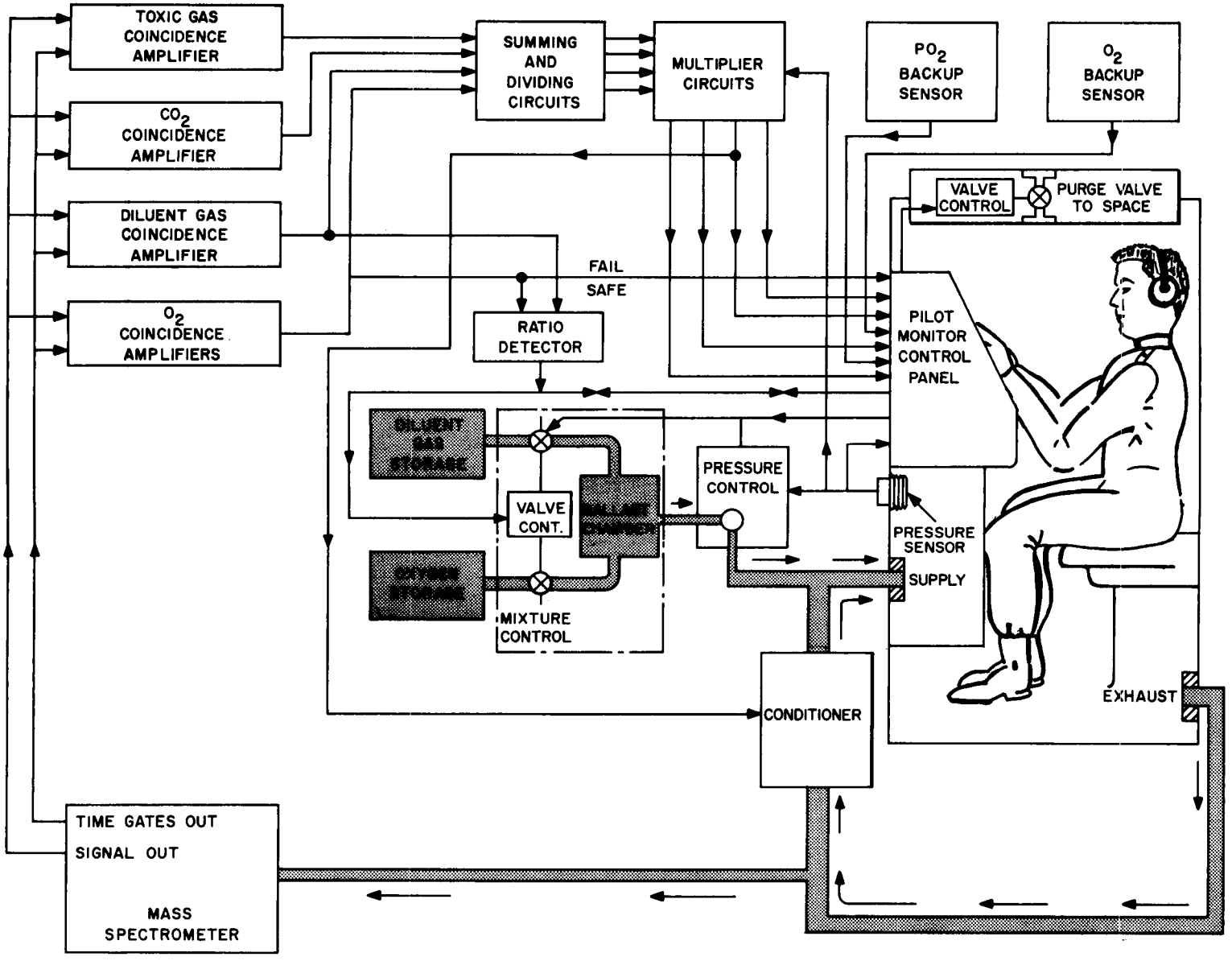


Figure XI-21-1. Use of mass spectrometer in a system for automatically maintaining balance of normal significant atmospheric constituents

CONFIDENTIAL

When a breakdown of the primary technique for monitoring and controlling the space cabin atmosphere occurs, it will become necessary for the crewmen to switch to override controlling of the atmosphere. The "fail-safe" indication characteristics of the mass spectrometer make possible the immediate warning that its determinations can no longer be depended upon. After a particular mass spectrometer has been calibrated for its mass 16 to mass 32 ratio (atomic oxygen to molecular oxygen) and if subsequently this ratio varies beyond well-defined limits, a warning signal will indicate that the mass spectrometer automatic system can no longer be depended upon. At this indication, the second mass spectrometer which has been in a stand-by condition can be phased into the system or the automatic system immediately abandoned and manual control assumed.

The cabin total pressure sensor will be a potentiometer-type aneroid with a high-level electrical output. Cabin wall temperature will be measured by a special surface-type thermistor operating in a simple resistive divider circuit. Atmospheric temperature will be measured by a bead thermistor configuration in a similar circuit. The mass spectrometer, when operating at its optimum reliability mode, is inherently inaccurate for partial H_2O determinations. For this reason, it will be necessary to monitor relative humidity making use of other standard reliable techniques.

INFRARED AND MAGNETIC SUSCEPTIBILITY TECHNIQUE

The infrared CO_2 monitoring instrument makes use of the infrared absorption characteristics of carbon dioxide to determine its presence and concentration. In the laboratory, these instruments have proven highly reliable and of high accuracy provided periodic calibration with known concentration gases is carried out. The output is a continuous electrical signal whose amplitude is a function of the partial pressure of the CO_2 in the atmosphere. It should be relatively easy to package the instrument for flight purposes by applying state-of-the-art miniaturization solid-state components. It is expected that the flight instrument will occupy approximately 0.5 cubic feet. The power required by this instrument should be no more than 20 watts. No "fail-safe" indication characteristic is exhibited by this instrument; failure can only be detected by abrupt failure of the output signal or by calibration.

The magnetic susceptibility technique is made use of in an instrument to provide an effective oxygen partial pressure sensor. Use is made in this technique of the paramagnetic characteristics of oxygen. A rotatable magnetic body is held in equilibrium in a magnetic field in opposition to the torque force applied by a quartz fiber. An increase in the partial pressure of oxygen passing around this rotatable body results in a lowered flux density through the body and, therefore, an unbalanced force. This unbalance is transduced as an electrical signal.

Packaging this instrument for flight may prove to be a problem. The problem area will be the fragility of the quartz fiber to transverse g forces. The manufacturer states that if the high g forces expected are (by instrument orientation) made to parallel to the quartz fiber axis this problem will be resolved. Here again, high accuracy can be maintained, provided the instrument is periodically calibrated with known oxygen concentration mixtures. The calibration plumbing would be provided as part of the package.

The flight package instrument should weigh not more than 12 pounds and occupy a volume of less than 0.6 cubic feet. Power requirements should be in the order of 20 watts. Failure of this instrument to provide accurate determinations can only be detected by complete loss of signal output or by calibration.

It can be seen from the block diagram showing this instrumentation utilized for sensing that the system is very similar to that employing the mass spectrometer. The back-up sensing system for atmosphere composition remains the same, as does the sensing for wall temperature, atmosphere temperature, and total cabin pressure. Also, all controlling functions remain identical. The major difference is the manner in which the instrument output signals are operated upon by the electronic circuitry for application to the controlling functions. For instance, in the mass spectrometer scheme the diluent gas controlling function is driven by a signal derived directly from the diluent partial pressure. In this scheme, the diluent gas control is driven by a signal which is the difference between the total cabin pressure and the sum of the carbon dioxide and oxygen partial pressures. Here differencing circuits are necessary, since only the partial pressure of oxygen and carbon dioxide are being sensed.

One major disadvantage of this system, other than the fact that periodic calibration is is required, will be the inability of this, the primary system, to monitor any other toxic or noxious gas or vapor regardless of its maximum acceptable concentration. With this system it will be mandatory that gas instrumentation for toxic and noxious gas determinations be carried.

GAS CHROMATOGRAPHY TECHNIQUE

Gas chromatography, with its requirements for a continuous carrier gas (helium) during analysis and the long delay time required to complete a single analysis, would appear to make it not acceptable as a primary sensing system for atmospheric monitoring. In any feedback system, in order to insure stability, it is desirable to continuously evaluate the error signal. Using the gas chromatograph, this obviously would be impossible. Storage techniques would be necessary; and even at best with a reasonably stable system, wide differentials would be experienced in the atmosphere composition.

COMPARISON OF TECHNIQUES

Table XI-21-1 compares characteristics of mass spectrometer, infrared and magnetic susceptibility, and polarographic approaches. As indicated, the mass spectrometer approach appears to be superior, both for present and future applications.

PRIORITY TELEMETERING OF GAS ANALYSIS DURING EMERGENCY

When the mass spectrometer is used as the sensor in the primary system for monitor and control of the cabin atmosphere, it becomes possible, electronically, to continually examine the output mass spectrum of the instrument for the presence of spectral peaks, indicating the presence of other than the known gas constituents designed into the system and those which it has been impossible to design out. The presence of any unknown peak of sufficient concentration can then be made to "commandeer" a high-resolution telemetry channel to effect analog transmission of the complete mass

TABLE XI-21-1. COMPARISON OF VARIOUS GAS-SENSING INSTRUMENTATION SYSTEMS

Characteristics	Total System*	Mass Spectrometer (omnigas)	Infrared and Magnetic Susceptibility (O_2 + CO_2)	Polarographic (O_2 + CO_2)
Packaged for Flight	1	Yes	No	Yes
	2	-	-	-
Can be Packaged for Flight	1	-	No	-
	2	-	Yes	-
Weight (Flight-Packaged)	1	18 lbs	25 lbs	4 lbs
	2	13	15 lbs	-
Volume (Flight-Packaged)	1	0.7 ft^3	1 ft^3	0.25 ft^3
	2	-	0.6 ft^3	-
Power Required	1	20 watts	45 watts	2 watts
	2	-	25 watts	-
Inflight Calibration Required	1	No	Yes	Yes
	2	-		
Single Instrument For Omni-Gas Sensor	1	Yes	No	No
	2	-	No	No
Reliability Longevity	1	Good	Fair	Fair
	2	Excellent	Good	Good
"Fail-Safe" Indication	1	Yes	No	No
	2	-	No	No
Atmosphere Lost	1	Negligible	None	Negligible
	2	-	-	-
	1			
	2			

*Includes all hardware necessary for generation of control signals suitable for operating controlling elements (valves, etc.).

1. Immediate hardware capability
2. Future hardware capability

spectrum for examination by competent spectroscopists on the ground. The same "commandeering" signal can also warn the crewmen of the presence of an unknown atmospheric constituent, permitting them, in the absence of instructions from the ground to take corrective measures, such as purging of the cabin, in an attempt to remove or lower the concentration of the unknown.

The block diagram of the system, Figure XI-21-2, shows the mass spectrometer signal output applied to the total analysis channel. Here it is separated, amplified, and converted to a suitable impedance level for sweeping by the analog strobe. Output from this circuit is a slow-speed scan of the complete mass spectrum for application to the extinction gating circuit and the telemetry electronic switch. The gating circuit removes all knowns and applies the unknowns, if any, to the threshold detector where any mass peaks above a preset threshold will cause an output switching signal—here, of course, integration will be used to prevent spurious triggering by noise pulses. In the presence of an output signal from the threshold detector, the telemetry electronic switch will switch the analog mass spectrum to the telemetry encoding circuitry and disconnect the data being "bumped." The "sum" signal from the mass spectrometer summing circuit is converted to an automatic gain control signal by the AGC amplifier and converter and applied as a threshold bias control to the threshold detector. This action adjusts the threshold detector gain to compensate for sensitivity changes in the mass spectrometer. Table XI-21-2 lists the weight, volume, and power requirements of the circuit components shown in Figure XI-21-2 and described above.

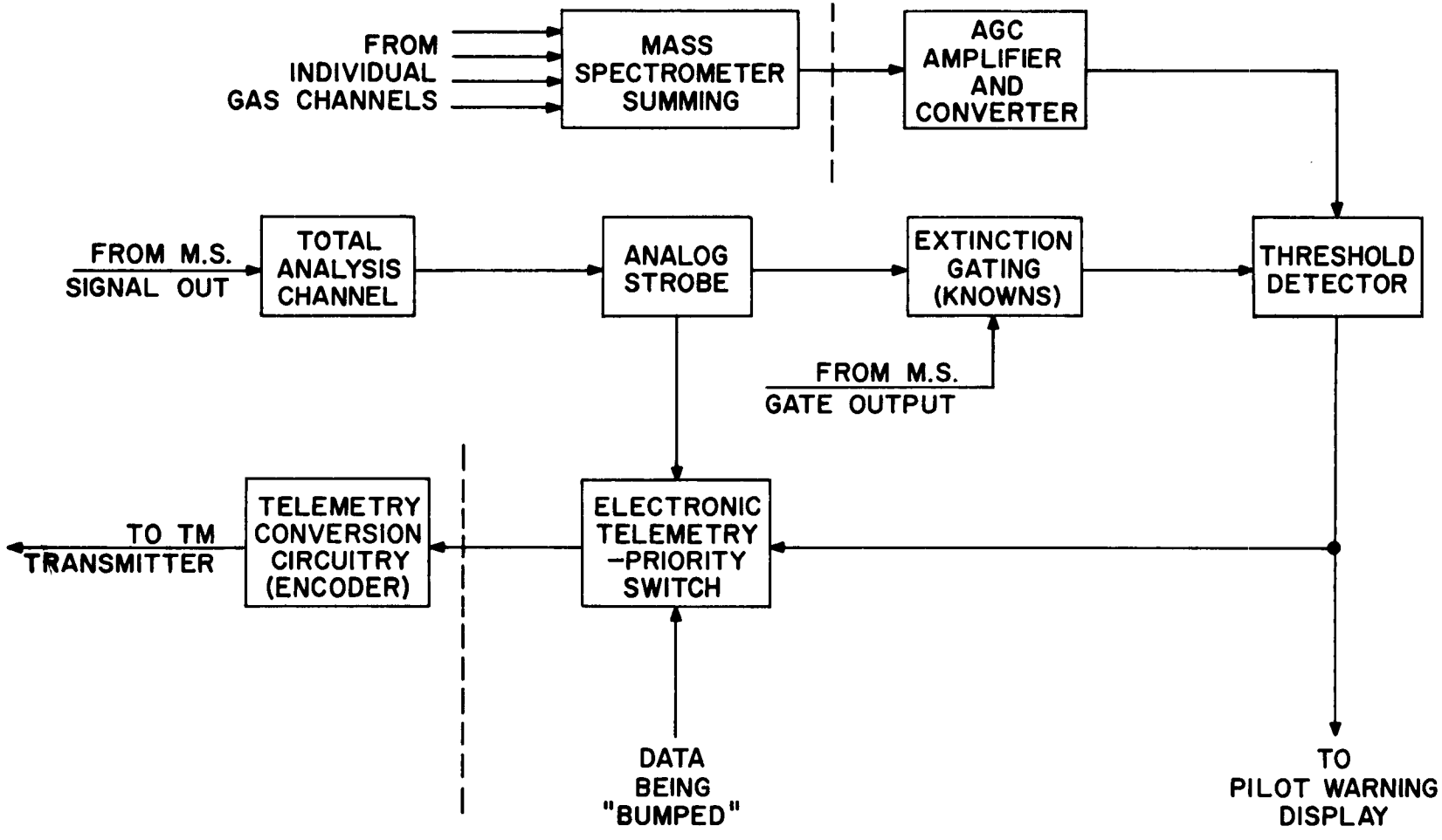


Figure XI-21-2. Priority telemetering of gas analysis during emergency, block diagram

TABLE XI-21-2. CIRCUIT COMPONENTS DATA

	Weight	Volume	Power Requirements
AGC Amplifier and converter	5 oz	6 $in.^3$ 2X3X1	0.25 watts
Total Analysis Channel	5 oz	6 $in.^3$ 2X3X1	0.15 watts
Analog Strobe	12 oz	15 $in.^3$ 3X5X1	0.5 watts
Extinction Gating	6 oz	6 $in.^3$ 2X3X1	0.15 watts
Threshold Detector	16 oz	20 $in.^3$ 4X5X1	0.5 watts
Electronic Telemetry Priority Switch	8 oz	10 $in.^3$ 4X2X1	0.3 watts
Totals	3 lbs, 4 oz	80 $in.^3$	2.00 watts

CHAPTER XII

INSTRUMENTATION & COMMUNICATIONS

XII. INSTRUMENTATION AND COMMUNICATIONS

1. INSTRUMENTATION

VEHICLE INSTRUMENTATION

Telemetry

The vehicle telemetry system employs pulse code modulation for the accumulation, multiplexing, coding, and transmission of some 448 data points. The selection of PCM was based on its high information efficiency in relation to transmission ranges, power limitations, and data requirements.

A block diagram of the primary airborne communications system is shown in Figure XII-1-1. The basic vehicle telemetry system is outlined in the lower half of this illustration.

The PCM encoder has an information bit rate of 53.76 KC. After the encoder output is converted to NRZ modulation, the pulse rate is 26.88 KC. The PCM format consists of 6 sub frames of 128 words each, a format compatible with existing PCM ground stations. Each word is defined by 6 digits (plus 1 for parity) yielding a quantization accuracy of 1.59 percent.

Consideration was given to the use of two PCM encoders operating at different bit rates and detecting different groups of sensors. However, PCM data capacity per unit bandwidth is so large that the increase in margin realized by the resulting reduction of the transmitted pulse rate would not offset the additional weight and complexity required. All data points, therefore, will be sampled, encoded, and transmitted at the nominal sampling rate. Data points of interest can be decoded and extracted from the PCM ground stations as required.

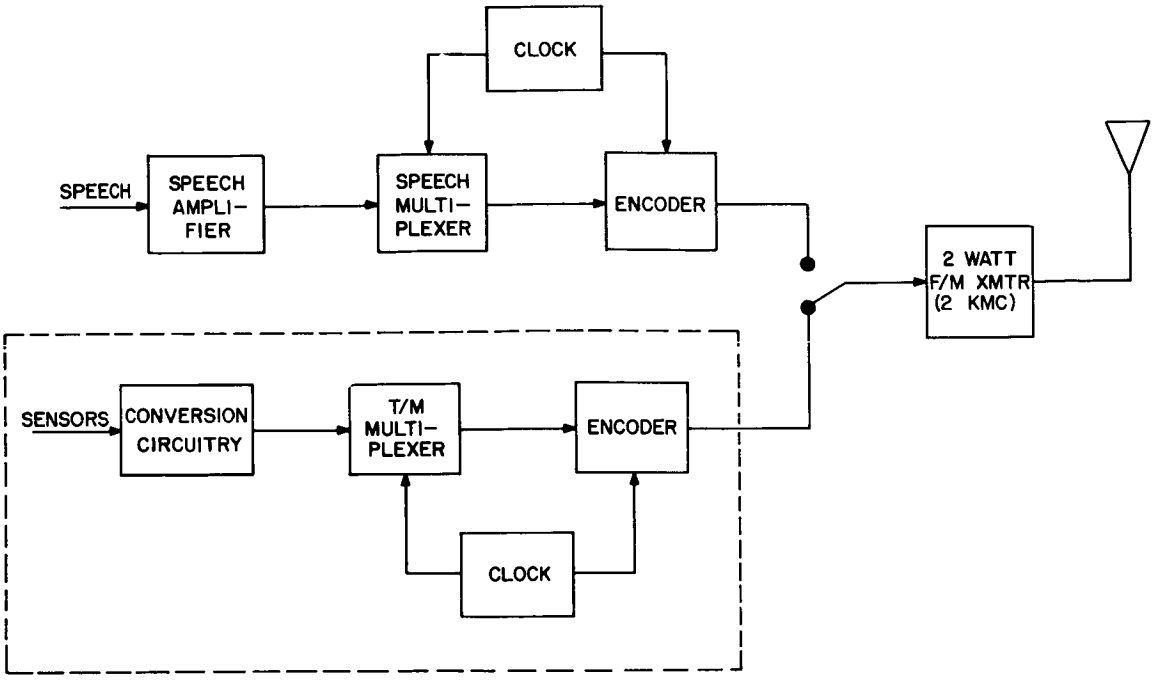


Figure XII-1-1. Block diagram of primary airborne communication system

The following tabulation represents the estimated data requirements for the telemetry system:

a. 2 points at 240 s/s
   2 - Biomedical: EKG

b. 50 points at 60 s/s
   38 - Aerodynamics
   4 - Biomedical
   3 - Navigation
   3 - Propulsion
   2 - Space Power

c. 420 points at 10 s/s
   26 - Aerodynamics
   50 - Biomedical
   70 - Navigation
   50 - Propulsion

50 - Space Power
50 - Vehicle
50 - Environmental
50 - Miscellaneous
24 - Frame count and synch

Some of the data sampling points; e.g., aerodynamics, can be either eliminated or replaced by other sampling points when the data from those points is no longer useful.

Television

In the event that television is employed for real-time transmission of information on the vehicle or lunar observation, the system will have the characteristics described below.

The television camera, generating a standard 525 line by 700 element raster, operates at a rate of 0.75 frames per second. This frame rate is achieved by slowly scanning the target plate of a long persistance image-orthicon, the face of which is exposed by a shutter at the desired frame rate. The video output signal consists of 283,500 effective elements (including synchronization) repeated once every 1.33 seconds.

The television system block diagram is shown in Figure XII-1-2. The diagram indicates that the video output from the camera is sampled and encoded by a PCM encoder operating at a 850.5 KC bit rate. After the encoder is converted to NRZ mod., the pulse rate is 425.5 KC. Each video element is defined by 4 digits yielding a quantization accuracy of 1 part in 15. Compression of the synchronizing pulses provides 2/15 of code bandwidth for synchronization leaving 13/15 of code bandwidth to define video intensity. Picture quality, therefore, is dependent on 13 discrete levels of grey.

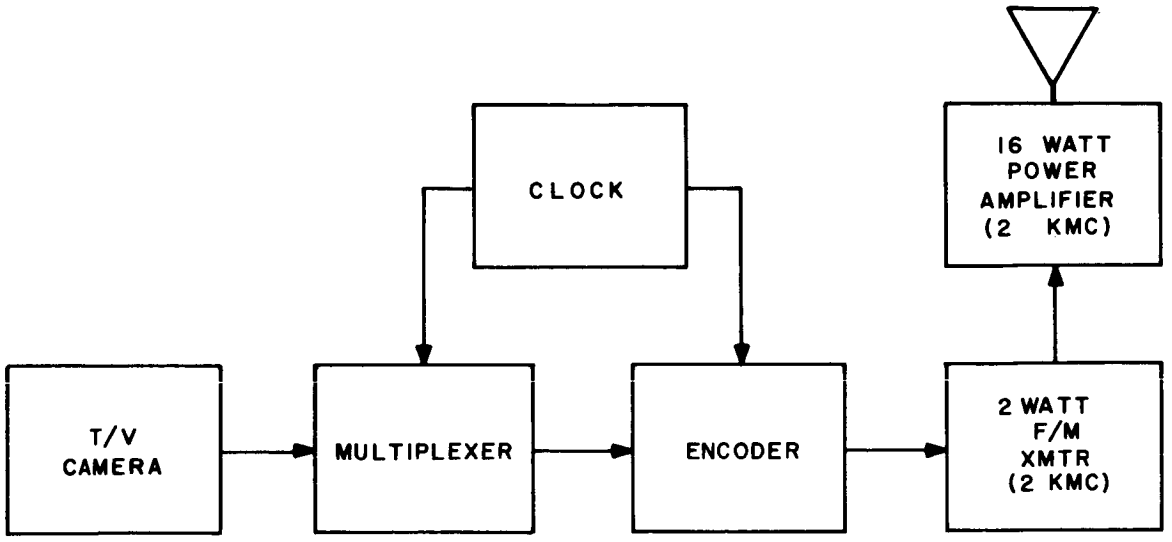


Figure XII-1-2. Block diagram of airborne television system

MISSION INSTRUMENTATION

A manned lunar-orbiting vehicle such as APOLLO can gather data on lunar topographic features and surface composition which will aid in selecting a site for an early manned-lunar landing.

A knowledge of the exact nature of the surface will be important in the determination of lunar landing subsystem requirements and the construction of a temporary lunar base

Prior to APOLLO, basic lunar environmental data will be obtained initially by means of probes, hard and soft-landing unmanned vehicles, and unmanned lunar satellites.

Probe and hard landing vehicles will be used to gather data on meteorites, gas densities, solar corpuscular radiation flux, solar radio wave emissives, cosmic radiation, ionization, magnetic fields and radiometric properties of the Moon.

Soft-landing vehicles will obtain some data on the lunar crust, surface temperatures and temperature gradients, seismic activity, natural radioactivity, and the gravity field at the Moon's surface.

Lunar satellites can carry instrumentation for meteorite flux measurements; detection of solar corpuscular radiation, solar ultraviolet radiation, cosmic radiation, magnetic fields, solar and lunar spectra and rough topographic data for initial mapping purposes.

Many of these experiments can be performed with high accuracy automatically, but some of them can be performed more accurately and efficiently if they are monitored by man.

The information gathered by a manned-observation vehicle can provide the more detailed, accurate data that is required to define the design requirements of an early lunar landing vehicle.

Of primary consideration in mission instrumentation is the determination of the equipment required to perform the observations essential to the location of a suitable landing site for a manned vehicle.

The description of a "suitable" landing site leads to the definition of the reconnaisance necessary to locate such a site and subsequently to the specification of the equipment and skills required to obtain the essential data.

The problem of site selection can be considered at two levels: macroscopic and microscopic. The macroscopic aspect will be concerned with the over-all "geographical" location; e.g., near side or far side of the Moon; east limb, center, or west limb; equator or polar, etc.

The microscopic aspect of the decision would consider whether the site should be located in a crater, bay, or mare; and whether it should be in the center of a large flat area or near a wall or mountain range. This latter aspect would be involved when preparing a preliminary layout of a lunar base.

A suitable landing site should have a flat surface consisting of high density material. It should be located in an area away from active hazards (radiation, seismic disturbance).

In view of the above restrictions, the reconnaisance necessary to detect a site that fits these conditions can be defined.

The following tabulation represents the minimum number of criteria necessary for the location of a suitable landing site:

a. Detection and storing of observation data concerning lunar topographical features.

b. Observational analysis of the structure and composition of the lunar crust.

c. Detection of possible active disturbances (radiation, seismic).

The detection and storing of data concerning lunar topographic features will be valuable in that it will provide information on surface features for use in a landing site location.

A digital terrain model of the area studied by APOLLO sensors will also aid in the guidance of the lunar landing vehicles. As a result of the APOLLO mission, there will be available a digital terrain model which reproduces, with certain accuracy limits, the three dimensional surface configuration of the overflight area. A lunar landing vehicle which approaches a portion of the lunar surface that has been mapped by APOLLO sensors will record, among other parameters, the topographic profile of a large area along the flight path. By matching the profile sensed by the landing vehicle with the digital terrain model, positional control of the vehicle and its flight path can be established.

At present the lunar surface detail is known to a relative accuracy of ± 1000 feet. By 1964, unmanned lunar satellites may provide a relative accuracy of lunar surface detail of ± 300 feet. The relative height of lunar surface features is known only to an accuracy of one-half mile. These accuracies are determined by sensor capabilities and position measurements.

The men in the APOLLO vehicle can provide the accurate geometrical measurements necessary to increase the accuracy of the mapping experiments as well as program and monitor these experiments.

There are several possible methods of obtaining lunar topographic data including radar, television, and optical-photographic techniques.

The problems associated with radar mapping systems are, briefly, high system weight, high power drain, and relatively low accuracy.

Television and optical-photographic systems have only one major problem: physical recovery of recorded information to yield high quality data. This problem is eliminated in the APOLLO mission because of the physical recovery module.

Optical-photographic systems will yield more accurate data (high picture quality) than known television observation systems for a mission of this type. It is this type of system that is pursued here.

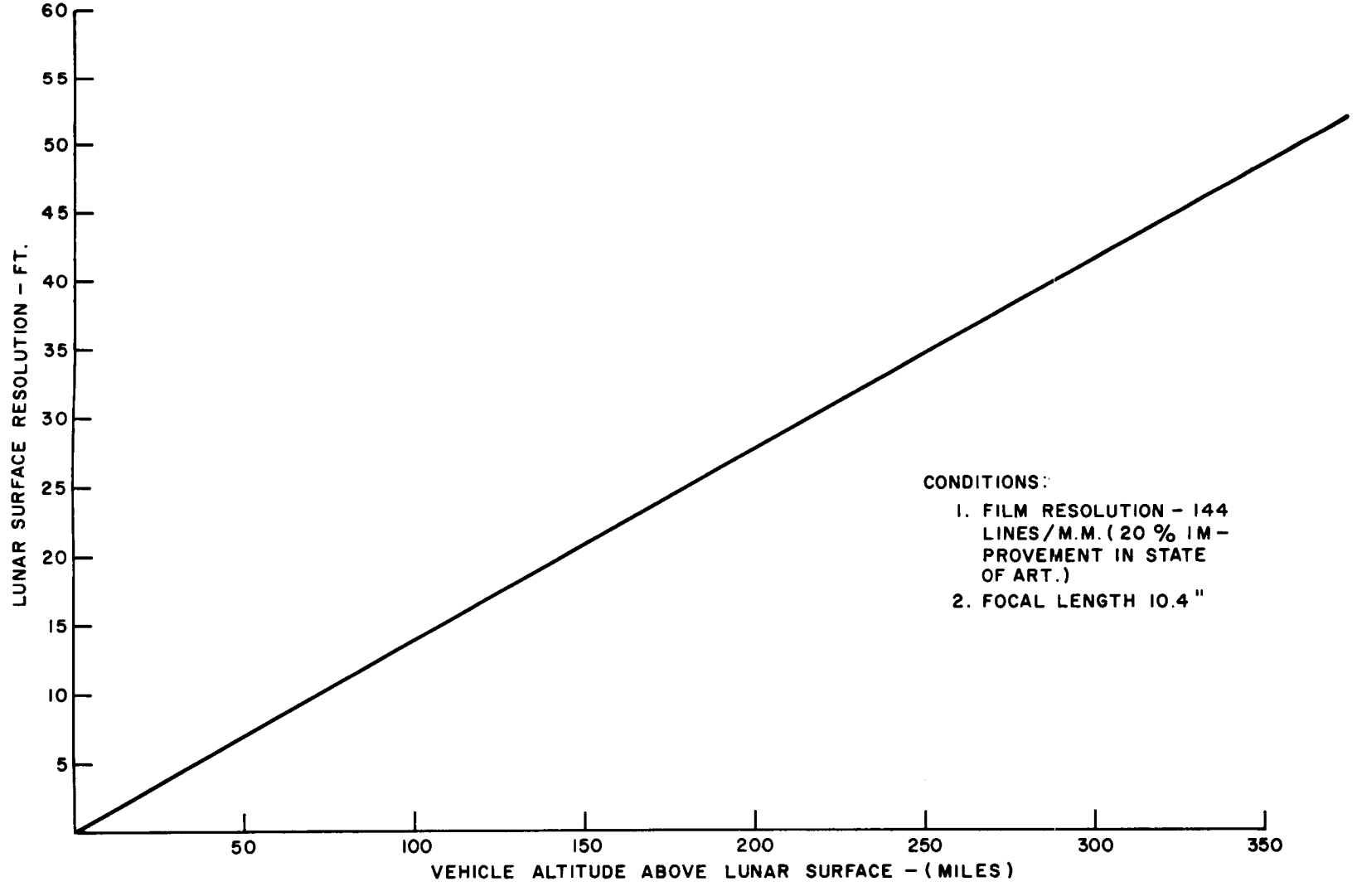
The lunar photographic observation system will consist of one automatically operated wide angle, scanning camera to be employed in obtaining data on general surface features over the entire visible sunlit portion of the Moon's surface; and one crew-operated narrow-angle camera for obtaining stereophotographic data on selected favorable lunar landing sights.

The wide-angle camera will have a field view of 51° and will execute a pre-programmed scanning motion so that the entire visible sunlit portion of the lunar surface can be photographed in a series of overlapping exposures.

For the nominal lunar orbit of 1000 miles apolune and 50 miles perilune, the maximum altitude at which photographs can be taken on the Moon's sunlit side will be approximately 380 miles with the perilune falling near the center of the sunlit area. In the range of altitudes between 50 and 380 miles the upper and lower limits of lunar surface resolution will be 7 feet and 53 feet respectively for vertical photographs; with 47 feet and 136 feet resolution for horizon photographs. A graph of lunar surface resolution versus vehicle altitude for the wide-angle camera is shown in Figure XII-1-3 (vertical photos only).

The camera will be equipped with a film transport that will roll the film automatically after each exposure. It will be designed to take photographs at a rate that will yield 10-percent overlap of vertical exposures, thus assuring complete lunar surface coverage at other angles of optical axis inclination.

CONFIDENTIAL



CONFIDENTIAL

Figure XII-1-3. Lunar surface resolution versus vehicle altitude for wide-angle camera (51.5°)

The wide-angle camera lens will be ten inches in diameter and approximately eleven inches long and will weigh about two pounds. The total weight of the camera including film, lens and transport is estimated at 15 pounds.

The picture taking sequence will be initiated and terminated by on-board command.

The narrow-angle camera will have a field of view of 4° and will be equipped with a telephoto lens for high resolution photography.

In its primary functional capacity, the narrow-angle camera will provide a means of obtaining detailed contour features of portions of the lunar surface that may be favorable for a lunar landing vehicle.

The areas toward which this camera will be directed will be determined by the crew. To obtain three-dimensional pictures of the selected areas, two photographs of each area must be obtained with a pre-determined interval between exposures.

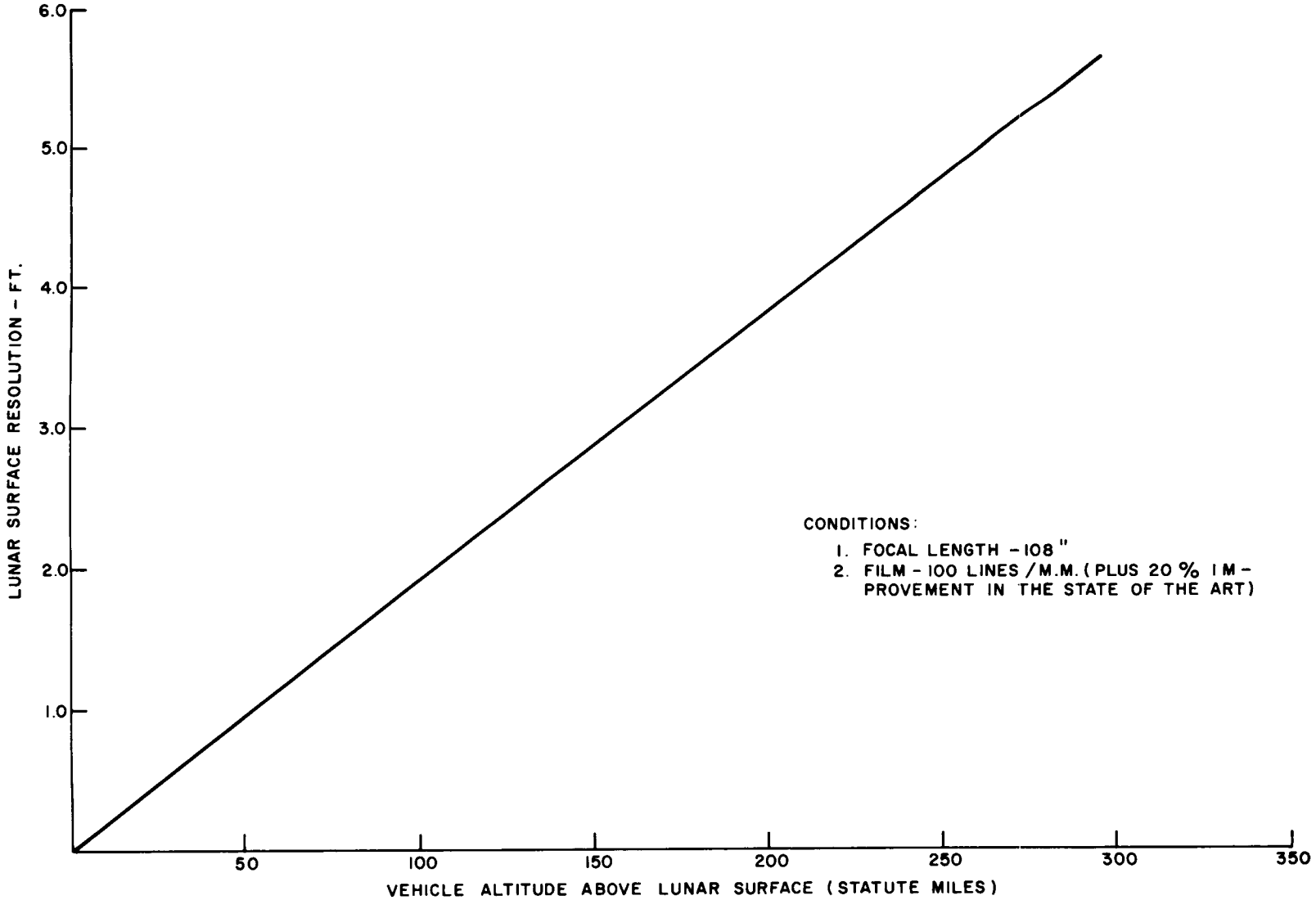
In order to obtain a measure of relative height of lunar surface features in the selected areas, several parameters must be measured at the instant the photographs are taken. These parameters are: vehicle altitude, vehicle position, and angle of inclination of the optical axis with respect to the local vertical. The interval between photographs of the same area must be measured and recorded so that the stereo base, which is required for calculation of relative height, can be determined.

The camera operator will initiate the recording of these parameters when he operates the shutter to photograph the selected areas.

The camera and film transport will weight about 25 pounds. Excluding the film transport, the camera will measure 26 inches in length and 8 inches in diameter. The effective focal length of the lens is 108 inches.

The combined film-lens resolution at the lunar surface is one foot at 50 miles and 7.3 feet at 380 miles (see Figure XII-1-4).

CONFIDENTIAL



CONFIDENTIAL

Figure XII-1-4. Lunar surface resolution versus altitude above lunar surface (narrow-angle camera)

The accuracy that can be expected in the determination of relative height as affected by camera limitations is 11 feet relative height error in a single measurement for exposures at an altitude of 50 miles and 84 feet for exposures at an altitude of 380 miles as shown in Figure XII-1-5.

Inaccuracy in the determination of the local vertical at the time photographs are taken will introduce an additional error (tilt error) in relative height measurements when the photographs are interpreted after the flight. The solid line in Figure XII-1-5 shows the total error in relative height due to resolution limits and a tilt error of 0.01°.

Radiation exposure of camera film can lead to additional errors in measurements.

If the camera film is exposed to radiation for extended periods, its resolution capability will be degraded. The film transport must therefore provide the necessary shielding to protect the film against the statistically anticipated radiation dose.

The equivalent of a one-centimeter thick aluminum film transport will keep the radiation dose at a level that will not seriously degrade the film resolution capability so that the desired topographic information can be gathered.

Photographs from the wide-angle camera will provide the information necessary for the construction of an accurate two-dimensional map of the lunar surface. Rough estimates of relative height can be obtained from shadow data with a knowledge of the angle of incidence of the sun's rays on the lunar surface at the time of film exposure.

Stereophotographs of selected areas will provide data on relative height of lunar surface features so that an accurate contour map of these areas can be constructed to aid in the preparation for an early lunar landing.

The second major experiment, the analysis of the structure and composition of the lunar crust, can provide detailed surface characteristics that no space-based camera system can detect.

It is desirable, for example, to know whether the surface is firm enough to support a vehicle which may land on it.

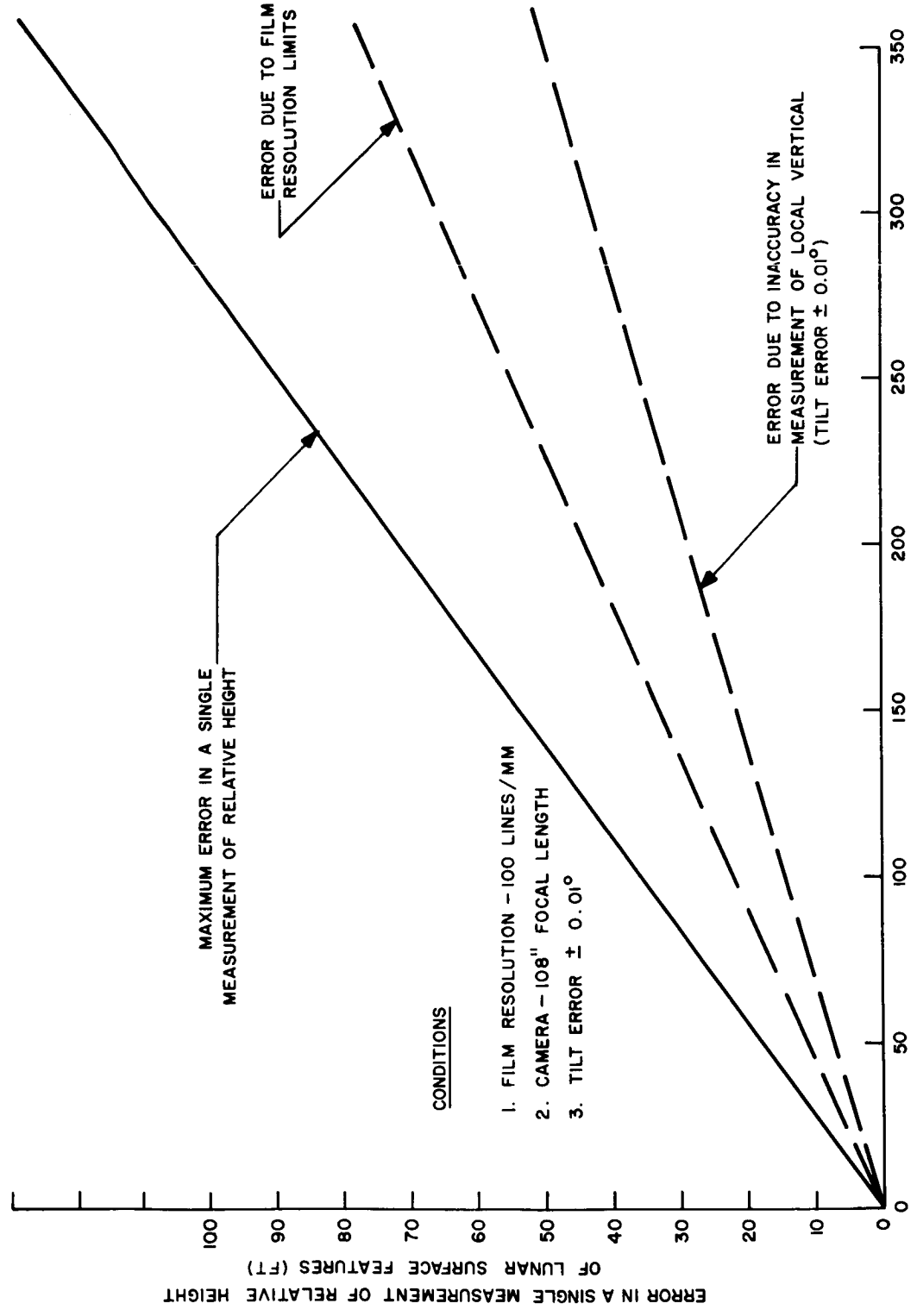


Figure XII-1-5. Maximum error in determination of relative height of lunar surface features versus vehicle altitude above lunar surface (narrow-angle camera)

The analysis of the structure of the lunar crust can be performed by spectral observations. In view of this, the spectrometer is being evaluated as a potential mission component. Its data-gathering capabilities are being considered in relation to mission purpose.

The spectrometer will detect spectral reflectance which is a measure of the instantaneous value of the ratio of the total reflected energy to the total incident energy on the lunar surface as a function of wavelength. A further definition of the type of reflectance; i.e., diffuse or specular, will lead to the determination of the relative smoothness or roughness of the surface in comparison to the range of wavelengths of the measurement spectrum.

The spectrometer will consist of four basic components: monochromator and detector unit, associated electronics; recorder, and power supply.

The spectrometer readings will lead to a detailed determination of surface texture to be used in conjunction with the stereoscopic photographs in selecting a most favorable landing site.

2. COMMUNICATIONS

The basic communications system requirements are as follows:

a. Continuous two-way voice link: (Figure XII-1-1)

b. Intermittent diagnostic telemetry and television: (Figures XII-1-1, XII-1-2)

c. A back-up communications link utilizing an omnidirectional vehicle antenna: (Figure XII-2-1)

It is anticipated that the existing and programmed capabilities of the NASA Deep Space Instrumentation Facility will be utilized for the APOLLO mission.

In the choice of operating frequencies, the factors of terrestrial noise, atmospheric attenuation, receiver capabilities, etc. were considered. These factors, plus the plans for the DSIF installations, led to a choice of approximately 2 KMC as the primary communications frequency for use in the preliminary configuration studies. For the back-up link shown in Figure XII-2-1, a frequency of 400 mc is utilized in order to obtain omnidirectional antenna pattern coverage.

Various modulation and transmission techniques were examined and compared in order to realize maximum margin for minimum complexity, weight, and power. Of the techniques considered, primary attention was given to PCM-AM, PCM-FM, and PCM-PS. All things being equal, PCM-FM generally yields about 2 db more margin than PCM-AM. While PCM-PS yields 6 db more margin PCM-FM, its somewhat greater complexity does not seem to justify its use in these applications. However, additional research and development on PCM-PS should result in system advantages applicable within the time scale of this program. For the present configuration studies, PCM-FM has been chosen as the primary means of vehicle to earth communications.

Voice communication is achieved by encoding companded speech audio on a 6-digit code, sampled at a 7-KC rate. This results in an encoder pulse rate of 21 KC NRZ_{mod}. The coded speech has an audio response of 3.5 KC with a resolution of 1 part in 63.

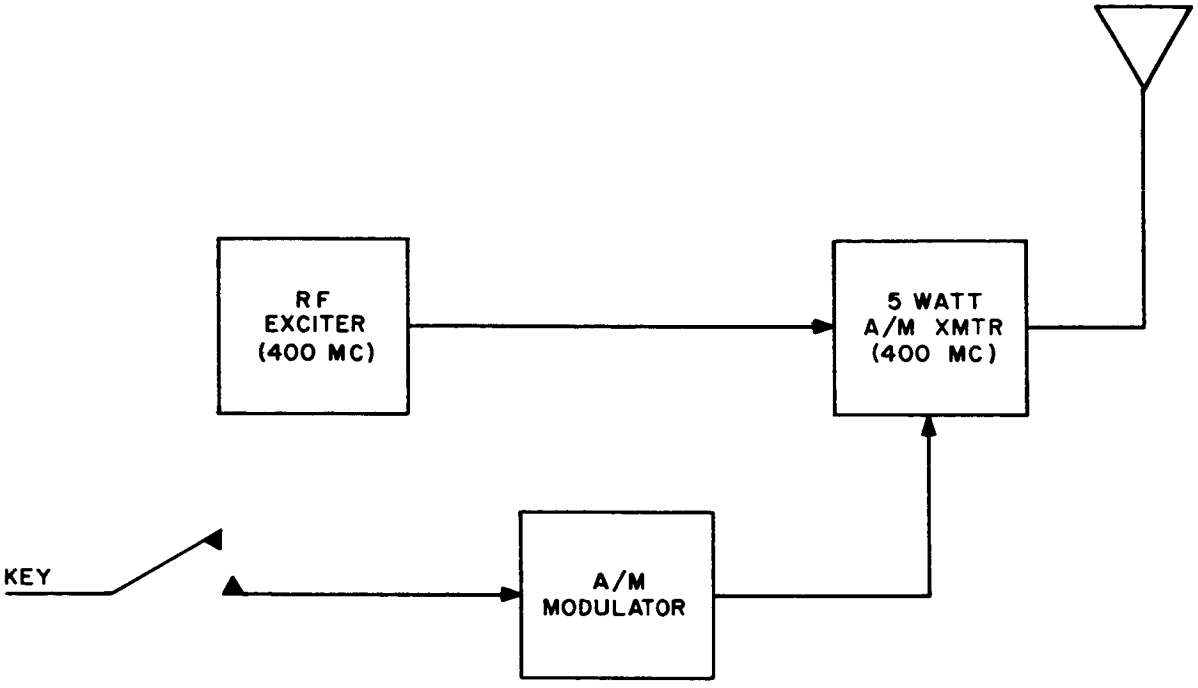


Figure XII-2-1. Block diagram of airborne emergency communications system

The primary communications link to Earth is effected by a directive antenna and transistorized 2-watt FM transmitter operating at 2 KMC. The transmitter is normally modulated by the 26.88 KC_{NRZ} output of the telemetry encoder yielding semi-continuous data coverage. When voice transmission is desired, the transmitter input is disconnected from the output of the telemetry encoder and connected to the output of the speech encoder. Thus, one time-shared r-f carrier serves for both telemetry and voice transmission.

The 2 KMC television carrier, also from the directive antenna, is generated by a transistorized 2-watt FM transmitter functioning as an exciter for a 16-watt power amplifier. The output of the video encoder is used to modulate the transmitter resulting in PCM-FM transmission of video. Due to the relatively high power required to operate the total television system, intermittant operation is planned.

The emergency communications link employs an omnidirectional antenna system and a transistorized 5-watt AM transmitter operating at 400 mc. Radio telegraphy modulation of the carrier is effected by manual operation of a key. Assuming that the key

operator is trained, a transmission rate of 20 words per minute is not an unreasonable rate.

Ground receiving equipment used employs low noise techniques in the design of the ultra-sensitive front ends. Masers and parametric amplifiers with noise figures of 1 db or less are necessary for this system. It is expected that considerable development is necessary to make this equipment a practical reality. Ground to vehicle voice communications is accomplished by standard FM techniques at both 2 KMC and 400 mc. The airborne FM receivers employ crystal mixer front ends. The graph in Figure XII-2-2 indicates that the noise figure for these receivers is 7db (or less) at the frequencies considered.

Table XII-2-1 indicates the margin calculations for the communications system based on the foregoing discussion of power levels.

Table XII-2-2 is a tabulation of the equipment requirements for the vehicle instrumentation and communication systems.

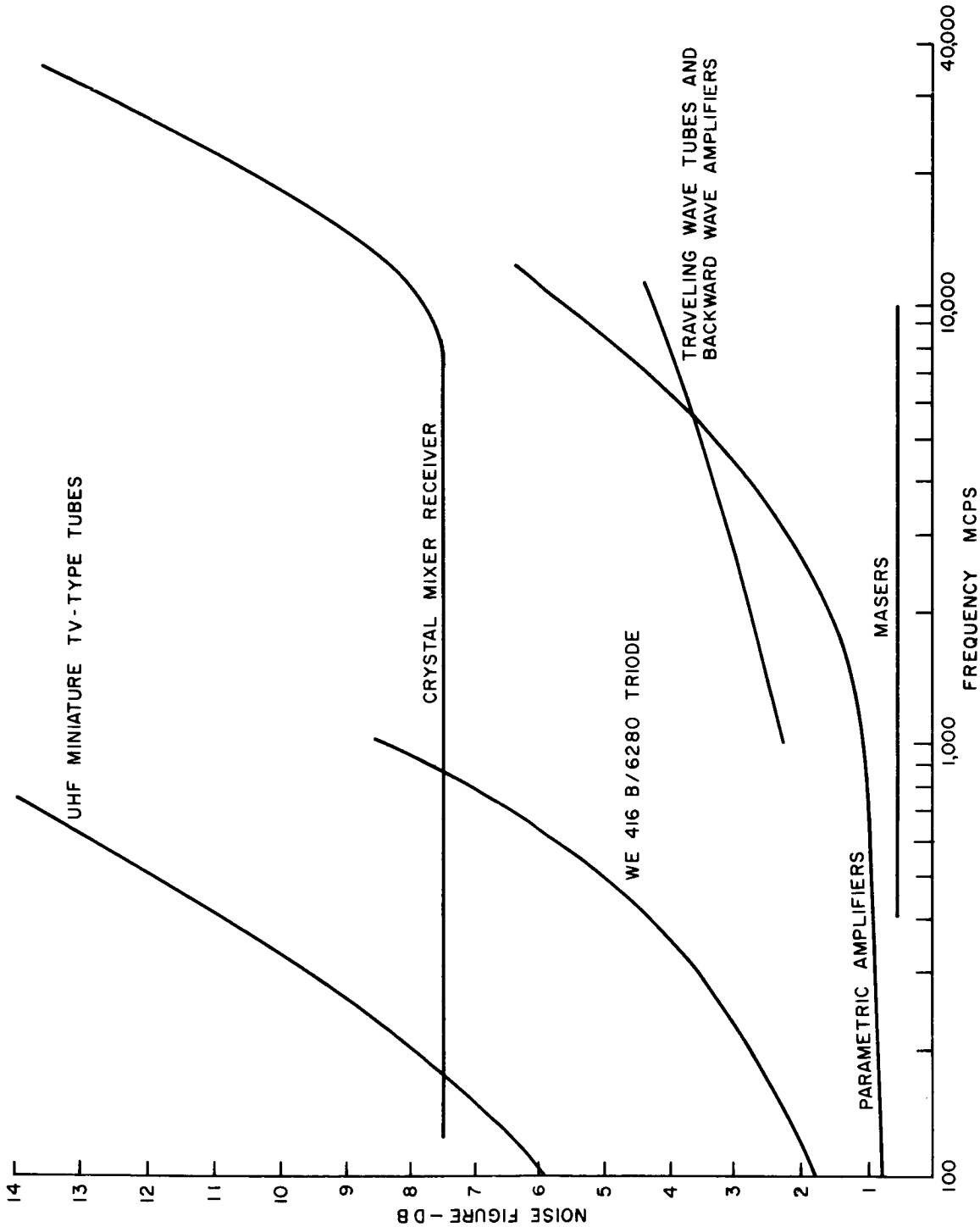


Figure XII-2-2. Noise figure of receiving devices

TABLE XII-2-1
MARGIN CALCULATIONS

Telemetry

$f_d = DF_c$

$= 0.6 \times 26.88$

$f_d = 16.128$ KC

$F_c = \frac{\eta M R}{2}$

$= \frac{768 \times 7 \times 10}{2}$

$F_c = 26.88$ KC

$BW = 3.4 f_d$

$= 3.4 \times 16.128$

$BW = 54.835$

KTB (100 KC receiver)	155	Space Attenuation	211
Receiving Antenna	50	Field Service Factor	6
Transmitting Antenna	25	S/N Required	8
Power Output	3	Receiver Noise	1
+ Total =	233 db	- Total =	226 db

Margin = 233-226

Margin = 7 db

Voice

$f_d = DF_c$

$= 0.6 \times 21$

$f_d = 12.6$ KC

$F_c = \frac{\eta M R}{2}$

$= \frac{1 \times 6 \times 7000}{2}$

$F_c = 21$ KC

TABLE XII-2-1 (Continued)
MARGIN CALCULATIONS

$BW = 3.4 f_d$

$= 3.4 \times 12.6$

$BW = 42.84$ KC

KTB (100-KC receiver)	155	Space Attenuation	211
Receiving Antenna	50	Field Service Factor	6
Transmitting Antenna	25	S/N Required	8
Power Output	3	Receiver Noise	1
+ Total =	233 db	- Total =	226 db

Margin = 233-226

Margin = 7 db

Television

$f_d = DF_c$

$= 0.6 \times 425.25$

$f_d = 255.153$ KC

$BW = 3.4 f_d$

$= 3.4 \times 255.153$

$BW = 867.52$ KC

$F_c = \frac{\eta M R}{2}$

$= \frac{283.500 \times 4 \times .75}{2}$

$F_c = 425.25$ KC

TABLE XII-2-1 (Continued)
MARGIN CALCULATIONS

KTB (1mc receiver)	144	Space Attenuation	211
Receiving Antenna	50	Field Service Factor	6
Transmitting Antenna	25	S/N Required	8
Power Output	12	Receiver Noise	1
+ Total =	231 db	- Total =	266 db

Margin = 231-226

Margin = 5 db

Emergency Communications

R = 25 words per minute

BW = 5B

= 5 x 20

BW = 100 cps

B = 0.8R

= 0.8 x 25

B = 20 characters per minute

KTB (200 cps receiver)	181	Space Attenuation	196
Receiving Antenna	36	Field Service Factor	6
Transmitting Antenna	-3	S/N Required	10
Power Output	7	Receiver Noise	4
+ Total =	221 db	- Total =	216 db

Margin = 221-216

Margin = 5 db

TABLE XII-2-2

EQUIPMENT DESCRIPTION

Component	Quantity	Weight (each)	Power (each)	Remarks
2-KMC, 16-Watt Power Amplifier	1	15	120	Intermittent operation
2-KMC, 2-Watt Transmitter	3	5	15	One continuous, One intermittent
400-mc, 5-Watt Transmitter	1	6	15	Emergency communications
Signal Conditioner	2	3	--	One in continuous operation
PCM Multiplexer/ Encoder	5	8	10	One in continuous operation
T/V Camera	1	17	100	Intermittent operation
2-KMC Receiver	2	4	8	One in continuous operation
400-mc Receiver	1	4	8	Emergency communications
Antenna Multiplexer	1	4	--	Deplexer/duplexer
Antennas	3		--	Three systems

CHAPTER XIII

ON-BOARD PROPULSION

XIII. APOLLO ON-BOARD PROPULSION

1. INTRODUCTION

The APOLLO on-board propulsion system may be logically divided into four main areas:

1. Abort
2. Mid-course correction
3. Lunar orbit and de-orbit
4. Attitude control

For the abort sub-system, primary attention has been given to solid rockets, although the propulsion team members (Aerojet and Thiokol) have been requested to examine a liquid system for comparison. The mid-course correction and lunar orbit and de-orbit systems have been examined for the use of solids, non-cryogenic liquids, and the high-energy liquid-oxygen — liquid-hydrogen and liquid-fluorine — liquid-hydrogen combinations.

Studies have been conducted for lunar orbit missions and basic circumlunar missions. The purpose of these studies, the results of which are described in subsequent sections of this chapter, was to obtain preliminary information on sub-system weights and volumes. As such, operational parameters as chamber pressure, expansion ratio, etc., are not necessarily optimum. Refinement of these studies is underway at Aerojet, Thiokol, and Bell Aerosystems to provide these optimum operating characteristics.

Also presented in this report is a summary of a conceptual approach to propulsion reliability. Because of the basic nature of the APOLLO mission, the ordinary standards and measures of reliability are not necessarily applicable, and a more fundamental approach to reliability attainment and maintenance appears warranted.

2. PROPULSION PHASES ANALYSIS

ABORT PROPULSION

Concurrently with the effort underway to specify the abort requirements, a preliminary parametric study has been made to enable a rapid evaluation of abort system weight to be made. Initial investigation has indicated that most stringent abort conditions exist for an on-the-pad abort and abort at maximum dynamic pressure. For on-the-pad abort, the vehicle must not only be rapidly removed from the vicinity of the booster, but must achieve sufficient altitude to permit deployment of a descent mechanism such as parachutes or rotor blades.

Utilizing simplified computational techniques, the data presented in Figures XIII-2-1 and 2 was generated. The first of these figures indicates the distance the vehicle has traversed from the booster as a function of the burning time of the motor. Results are shown for specific impulse of 220 seconds and 245 seconds performance at sea level. These values are representative of the older jato-type units and presently available propellants respectively. The propellants in these categories are extremely reliable and a large storehouse of experience is available. Higher energy propellants are available, but their potential weight savings are small at the expense of overall reliability. An additional parameter given in Figure XIII-2-1 is the initial thrust-to-weight ratio. Shown in Figure XIII-2-2 is the apogee altitude attained as a function of propellant burning time. The data are presented for a specific impulse of 220 seconds, but are applicable for 245 seconds with only small deviations. The simplified calculation technique used herein does not consider drag and thus presents optimistic results. As choices are narrowed down, more refined approaches will be applied, which will include not only drag, but also thrust vector angle as parameters.

Finally, in Figure XIII-2-3, a plot of propulsion system weight as a function of burning time is given. The data shown are for an abort system that will abort a 15,000 pound payload. For lower-weight vehicles, such as a command module, the propulsion system weight may be approximated by multiplying the weight of Figure XIII-2-3 by the ratio of the aborted weight to 15,000 pounds. Several values of propellant mass

CONFIDENTIAL

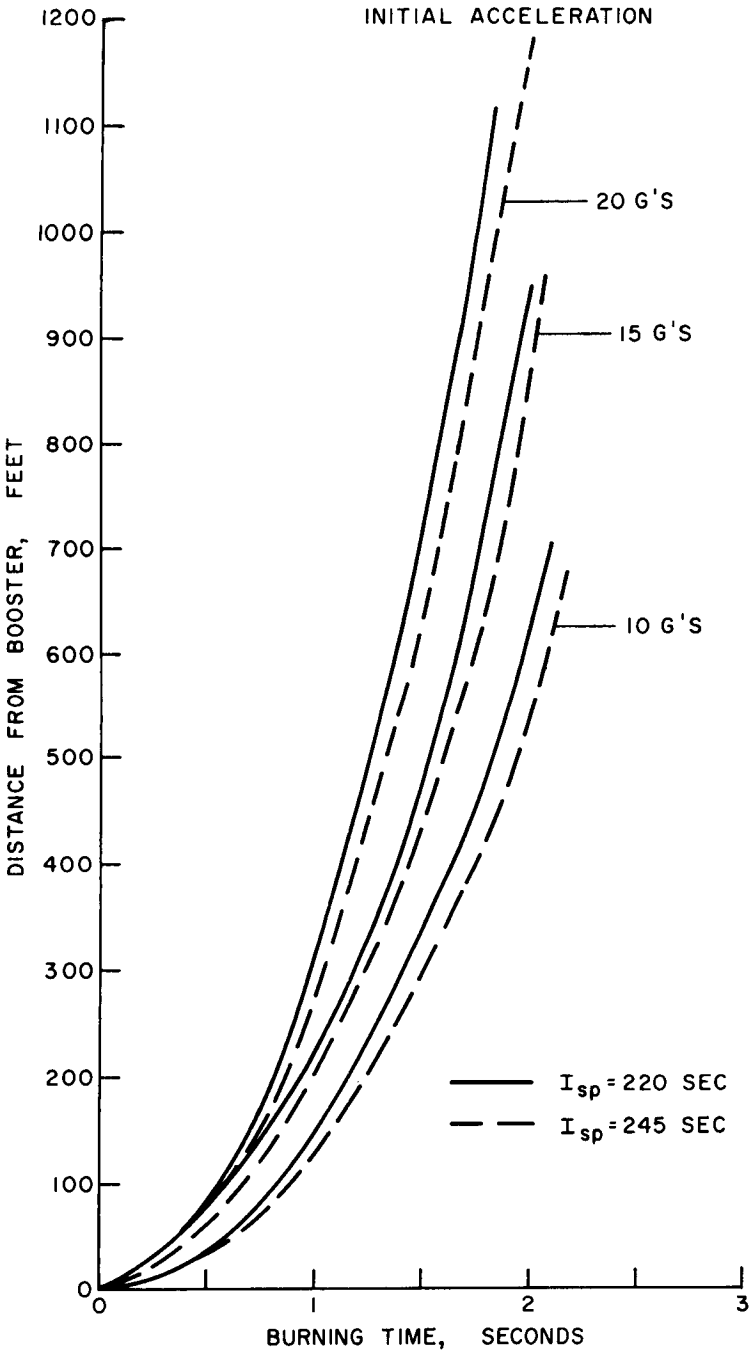


Figure XIII-2-1. Distance versus time, abort propulsion

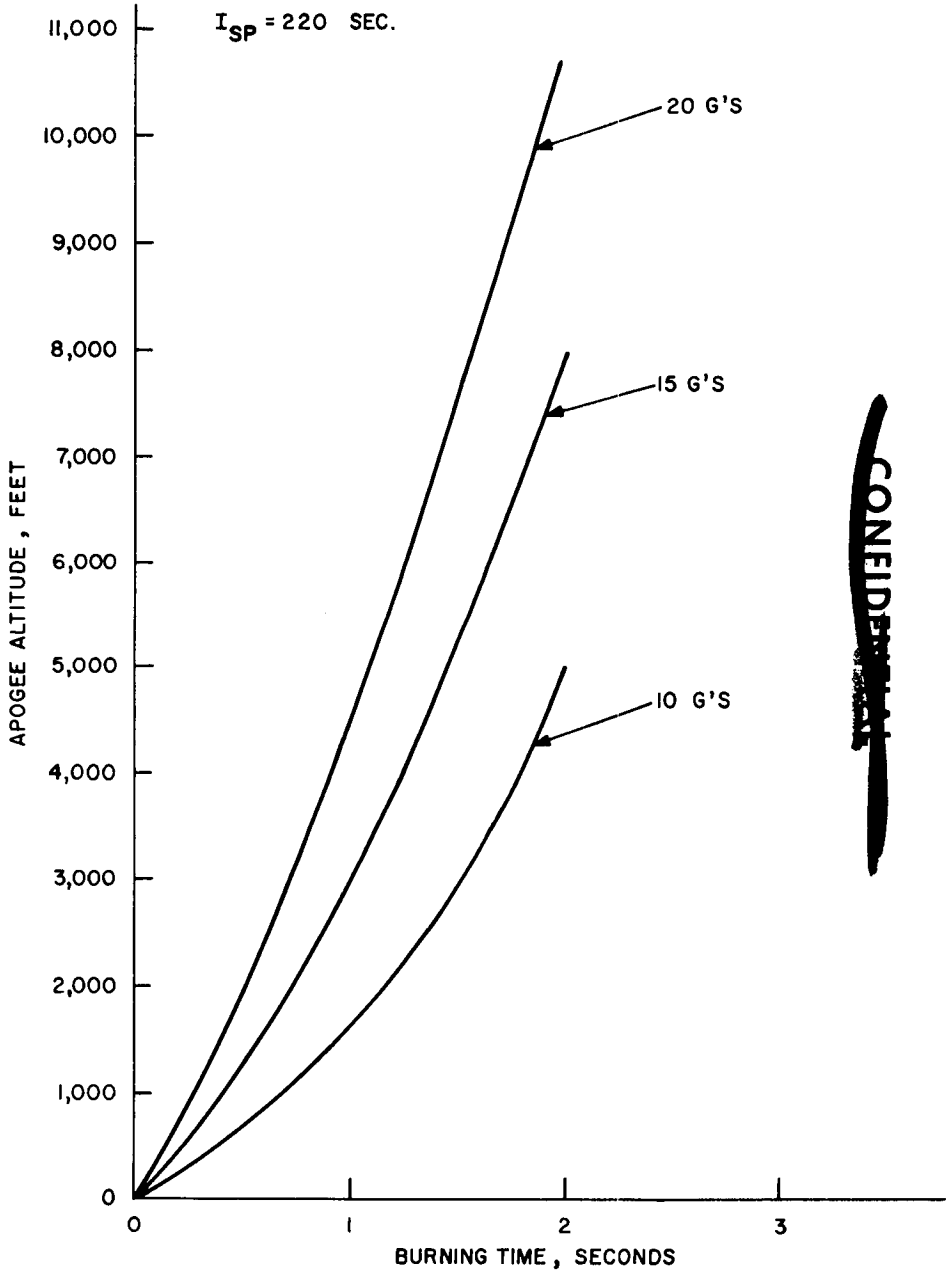


Figure XIII-2-2. Apogee altitude versus burning time, abort propulsion

CONFIDENTIAL

fraction (propellant weight divided by the sum of propellant weight and inert parts weight) are indicated. The actual value of the mass fraction is a function of the design refinement, material, safety factor, and the size of the unit (larger values of propellant weight will result in higher attainable mass fractions).

There are several important considerations that must be given further study to arrive at the number of abort rockets to be used. For example, with a single unit, a malfunction would have dire consequences and the abort system reliability will be that of the single unit. For a multiple-unit system, the probability of all units igniting is equivalent to the reliability of a single unit raised to a power equivalent to the number of units in use. For example, if the single-unit reliability were equal to 0.995, the probability of 10 units performing satisfactorily is $(0.995)^{10} = 0.951$ (see Figure XIII-4-1). However, if one or more of these units are redundant, barring a destructive malfunction, the overall system reliability approaches that of a single unit, and a failure of one unit will still permit a successful abort.

An additional advantage that may accrue from multiple units is that some of these may be discarded after the region of maximum dynamic pressure has been reached. Any weight that is dropped prior to ignition of the second and third stage of the booster will subtract only a few percent of that weight from useful payload that may be powered to escape velocity.

Preliminary calculations for on-the-pad abort assumed that the aborted vehicle should be at least 100 feet from the booster in one second, and it should attain an apogee altitude of at least 3000 feet. Observation of Figures XIII-2-1 and 2 for a specific impulse of 220 seconds indicates that an initial acceleration of 15 g is required with a burning time of one second. The requirement, in this case, is determined by the 3000 feet altitude. Assuming a typical multi-unit system with a conservative mass fraction of 0.6, the abort system weight will be found to be 1680 pounds in Figure XIII-2-3. The comparable figure for aborting only a command module is approximately 850 pounds. Improved case design and a 245-second specific impulse can reduce this weight to about 600 pounds. It should be stated that the mass fraction used is typical of jato-type units and may easily be increased by suitable redesign.

In Figure XIII-2-4, a plot of distance from the booster as a function of burning time is shown for three values of drag. This data is typical of conditions in the region of maximum dynamic pressure during boost. For a typical value of drag equal to 25,000 pounds (low drag fairing), it may be observed that the vehicle will be over 100 feet from the booster in one second for an initial acceleration of 15 g and a burning time of one second.

For aborted missions occurring at high velocities and altitudes, it will be necessary to utilize the on-board propulsion to redirect the vehicle in such a way that proper reentry angles and/or foreshortened flight paths will result. Preliminary study in this area discussed in Chapter VII using trajectory analyses to match requirements with energy availability, indicates that the on-board propulsion is sufficient to abort the mission safely over the entire trajectory.

MID-COURSE CORRECTION

The selection of the propulsion system for mid-course correction for the lunar orbiting vehicle may be dependent upon the system chosen for lunar orbit and disorbit. There exists the possibility, for liquid propellants, of utilizing the same tankage for the two systems and possibly using the same thrust chamber or chambers.

For the purposes of this preliminary study, a total mid-course correction of 300 feet per second was chosen. As many as eight impulses have been considered for providing the necessary corrections.

For a hypergolic bi-liquid propellant combination, this is a comparatively simple on-off type of operation. Utilizing a propellant combination of liquid oxygen and liquid hydrogen, a multiple ignition system must be utilized. Multiple ignitions for this combination will have been accomplished in the course of the Centaur Program and the knowledge will be applicable for APOLLO.

With a solid-propellant correction system, a multiple-unit system will be necessary. While the actual selection of required units is dependent upon the accuracy requirements and would necessitate a more detailed study, it was assumed that the mid-course

CONFIDENTIAL

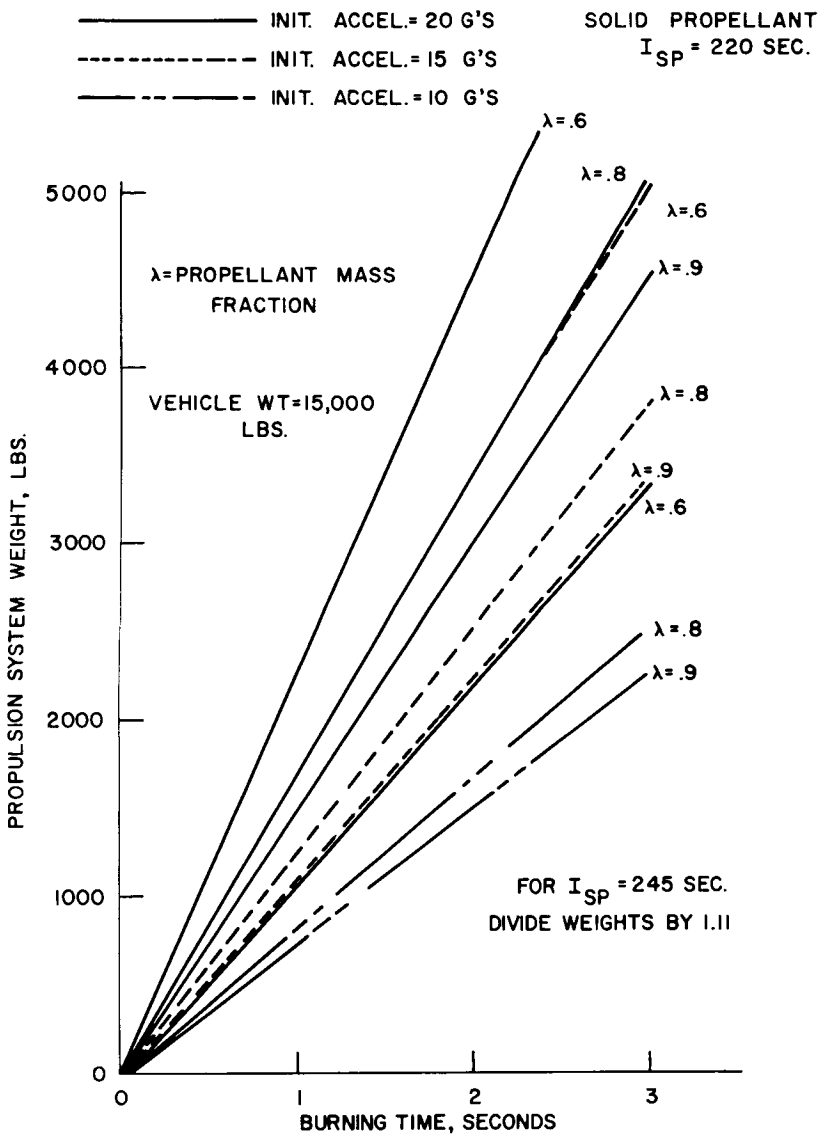


Figure XIII-2-3. Propulsion system weight for abort

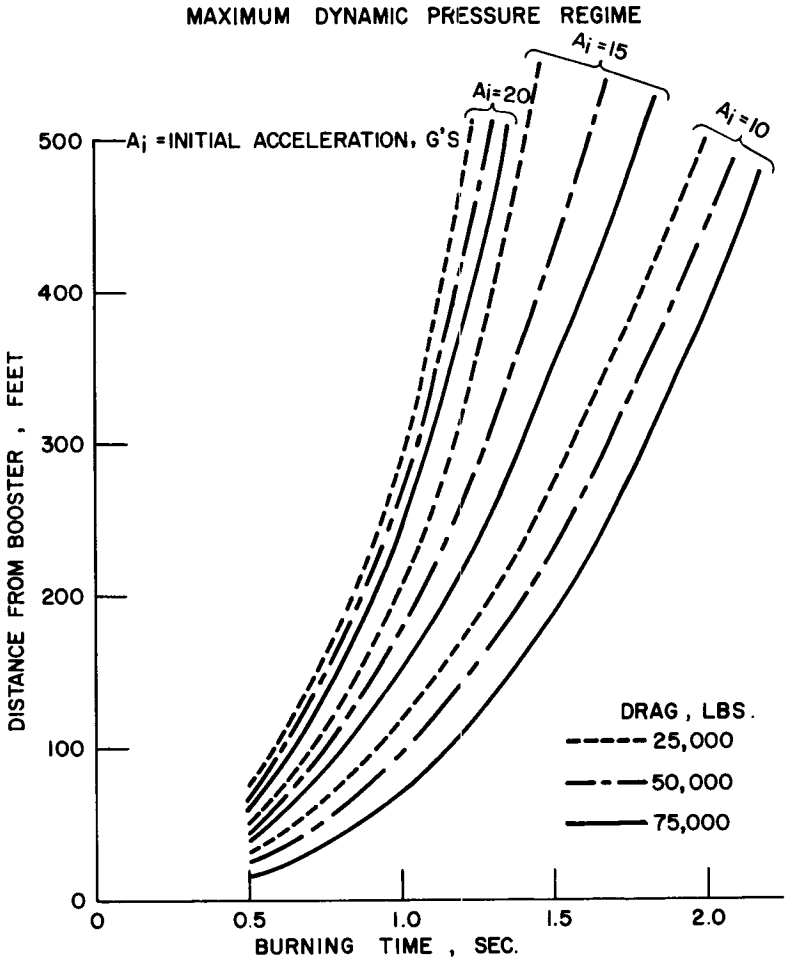


Figure XIII-2-4. Distance from booster versus burnout time

CONFIDENTIAL

correction may be provided by the units listed in Table XIII-2-1. The assumed mass fractions are shown and the total motor weight is given. For propulsion study purposes, it was assumed that half the units will be utilized for the outbound portion of the flight, and the remainder for the inbound portion. In addition, it was assumed that the velocity increment must be provided to the outbound vehicle weight. This is to ensure sufficient correction capability for the case in which a lunar orbit is not attempted. For solid propellants, a typical vacuum specific-impulse of 282 seconds has been utilized.

TABLE XIII-2-1

Velocity Increment, FPS	Number of Units	Propellant Mass Fraction	Unit Wt., Lbs.	Total Wt., Lbs.
5	10	0.75	11	110
25	4	0.85	50	200
50	4	0.88	95	380
Total solid propellant mid-course correction weight, pounds				690

The weight of a typical propulsion system for mid-course correction utilizing liquid propellants has also been obtained. This weight is included in total propulsion system weight shown as a function of incremental velocity in Figure XIII-2-5 for the storable propellant combination of monomethyl hydrazine and nitrogen tetroxide and for the higher-energy cyrogenic combination of liquid oxygen and liquid hydrogen. The specific impulses utilized for these propellants were 306 and 410 seconds respectively. It was further assumed that a pressurized system was utilized.

LUNER ORBIT AND ORBIT ESCAPE

For liquid propulsion systems, the weight of the propulsion system for lunar orbit and orbit escape is included in the total propulsion system weight shown as a function of total incremental velocity in Figure XIII-2-5. Various vehicle weights, prior to entry into the lunar orbit, are given to aid in rapid analysis. The initial vehicle weight is a function of the abort system weight and the manner in which the abort propulsion is dropped at various phases of the booster trajectory. In Figure XIII-2-6, a comparable

CONFIDENTIAL

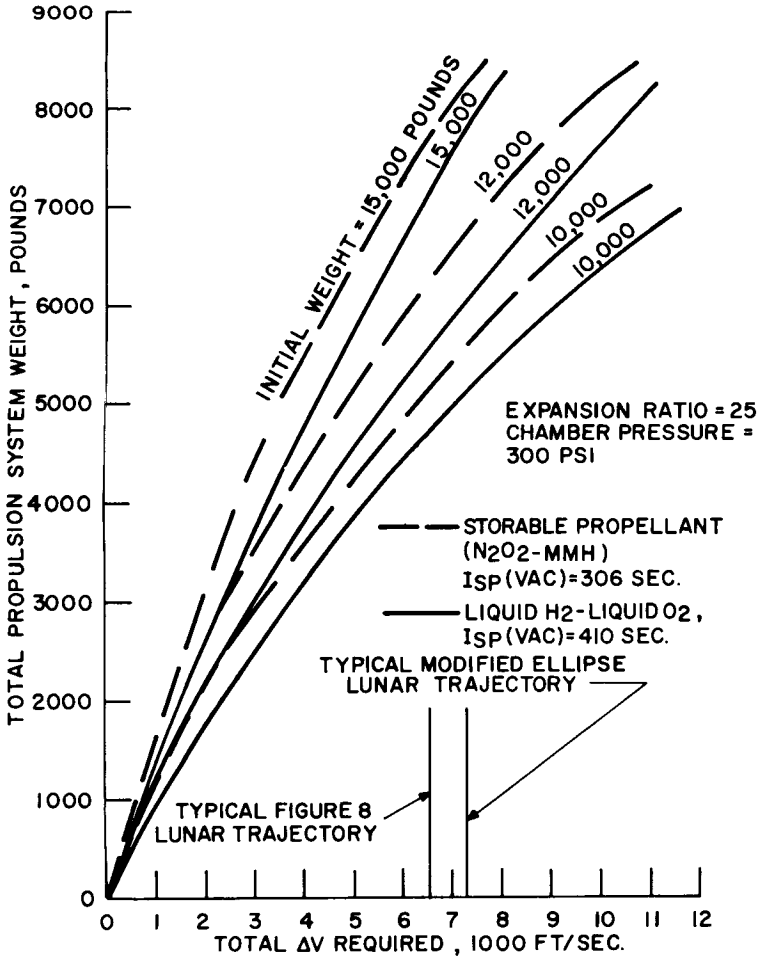


Figure XIII-2-5. Liquid propulsion weight for mid-course correction and lunar orbit and orbit escape

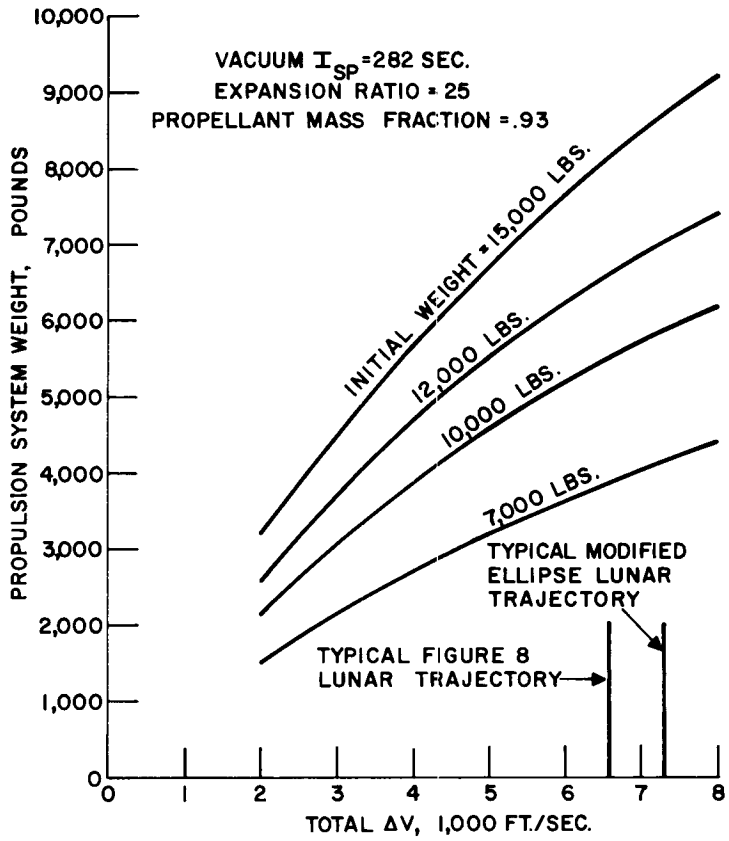


Figure XIII-2-6. Solid propulsion system weight for lunar orbit and orbit escape

CONFIDENTIAL

curve is shown for a solid propellant system. The use of these curves will be shown in the section titled Mission Propulsion Analysis. The basic assumptions made in obtaining these curves are similar to those used in the mid-course correction section.

ATTITUDE CONTROL

The studies to date have indicated that the energy requirements are low enough to utilize a separate system operating on storable hypergolic propellants. Since a large number of starts may be required, the propellant acquisition problem may be readily solved by the use of bladders in the tank. The hypergolicity of the propellants will require no external power for ignition and will utilize thoroughly proven technology. Data on the attitude control system will be found in Chapter VIII.

3. MISSION PROPULSION ANALYSIS

LUNAR ORBIT MISSION

For this mission, it has been assumed that sufficient abort propulsion has been carried to "save" a command module. Based on an estimated weight of 7000 pounds for this module, the space vehicle weight at escape velocity will be 15,000 minus 850 or 14,150 pounds.

Solid Propellant System

Assuming a total of 300 feet per second required for mid-course correction, the mid-course correction system weight was found to be 690 pounds in Table XIII-2-I of Section 2.

It was further assumed that the outbound mid-course correction units will not be dropped if they are not utilized in full or part. This will give a larger factor of safety on the return flight. Thus, the vehicle weight is 14,150 pounds, of which 690 pounds is for mid-course correction propulsion.

For a typical modified ellipse approach to the Moon, 3500 feet per second will be required for entry into the lunar orbit and the same velocity increment for escaping the orbit. From Figure XIII-2-6, a propulsion system weight of 4800 pounds will be required for injection into the lunar orbit. With a propellant mass fraction of 0.93, the burnout weight is approximately 300 pounds.

After entry into the lunar orbit, the empty case may be discarded and the vehicle weight becomes 14,150 minus 4800 or 9350 pounds. Utilizing Figure XIII-2-6 again with the new value of weight, the propulsion system weight for disorbiting is equal to 3200 pounds. Subtracting this figure and the mid-course correction weight of 690 pounds yields an allowable vehicle weight of 5460 pounds for a solid propulsion system. The small size of the solid propellant units results in low mass fractions. Further, the need for multiple sizes for a solid system may add significant development costs.

Storable Bi-Liquid Propellants

Utilizing the same velocity requirements as for the solid propellant system, the mid-course correction system weight may be found from Figure XIII-2-5 as approximately 500 pounds. When the values of chamber pressure and expansion ratio are optimized, it may be possible to achieve a weight reduction. This system, employing monomethyl hydrazine and nitrogen tetroxide, is lighter than the solid-propellant system. A higher specific impulse is utilized and no excess propellant is required since on-off operation is feasible.

Because of the need to assure sufficient propellant for the return phase of the mid-course correction, it may be advisable to operate the mid-course system from tankage separate from the propellant supply for the orbiting and orbit escape source. Further analysis is needed in this area.

For the lunar orbit and orbit escape system, requiring a total incremental velocity of 7000 feet per second, the propulsion system weight was found from Figure XIII-2-5 as 7550 pounds. The permissible vehicle weight is thus, 14,150 minus 500 minus 7550 = 6100 pounds. This represents an increase in allowable weight of 640 pounds compared to the solid propellant system.

Liquid Hydogen-Liquid Oxygen Propellants

The final system presented here employs liquid hydrogen and liquid oxygen for the mid-course and lunar orbit and orbit escape phases. From Figure XIII-2-5, the mid-course correction weight is found to be approximately 425 pounds. For the lunar orbit and orbit escape phases, a propellant system weight of 7100 pounds is required as also shown in this Figure. Finally, the vehicle weight may be obtained as 14,150 minus 425 minus 7100 or 6625 pounds. The hydrogen-oxygen combination thus will increase the allowable vehicle weight 525 pounds above a storable system and 1165 pounds above one utilizing solid propellants.

Summary Data

In Table XIII-3-1, a summary of the pertinent data is shown for the three systems. Further data is given in Figure XIII-3-1 to show the effect of variations in velocity

increment on allowable vehicle weight. The work to date, as summarized in Table XIII-3-1, has been somewhat conservative. Shown in Table XIII-3-2 are the results of an analysis using more optimistic assumptions typical of what may be available by 1966. Data is also presented for a liquid-hydrogen — liquid-fluorine system. This combination represents the maximum performance chemical propellant envisioned in the next ten years. For each propellant combination, maximum performance has been used with optimistic values of propulsion weight. The abort propulsion has also been modified to include high-performance propellants and lighter-weight cases. The results of this study indicate that the fluorine system will save approximately 400 pounds of weight compared to the liquid-oxygen — liquid-hydrogen combination. This weight savings must be tempered with the storage and handling problems of fluorine. Additional trade-offs are required to determine its usefulness in a manned system. Also indicated in parentheses in Tables XIII-3-1 and 2 are the allowable vehicle weights for a typical "figure-8" trajectory which utilizes a total mission velocity of 6600 feet per second.

In Figure XIII-3-1, a plot of ideal weight versus total mission velocity (sum of mid-course correction and orbit and disorbit velocity increments) is shown. The ideal weight is defined as 15,000 pounds minus the weight of propellant. Thus, the ideal weight includes abort propulsion weight, propellant hardware, vehicle structure, life-support, etc. This curve allows quick appraisal of the effects of mission velocity increment, and propellant combinations on the allowable weight of the remainder of the vehicle. Data are shown for solid propellant, a storable, liquid-hydrogen — liquid-oxygen, and liquid-hydrogen — liquid fluorine systems.

Weight Analysis

In Table XIII-3-3, a weight estimate and operating parameters are given for an O_2/H_2 system, and an F_2/H_2 system. The information is given for pressure fed systems. Two thrust levels are shown — a 2 g maximum thrust capability and a 1.5-g maximum thrust capability. The weights are predicated upon a 16,500 pound vehicle for the O_2/H_2 combination and 16,000 pounds for F_2/H_2. This data is typical of systems that might be available in the 1963 time period. This data was taken from material prepared by the Aerojet-General Corporation's Liquid Rocket plant. A typical curve is shown in Figure XIII-3-3 for an O_2/H_2 system.

CONFIDENTIAL

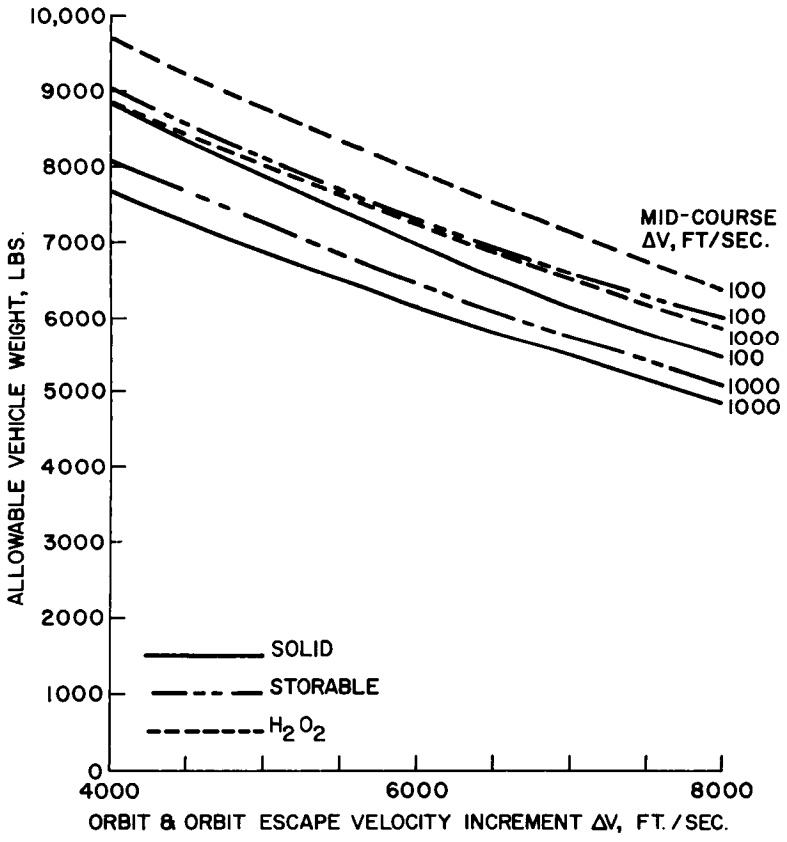


Figure XIII-3-1. Lunar orbit mission; vehicle weight minus propulsion weight

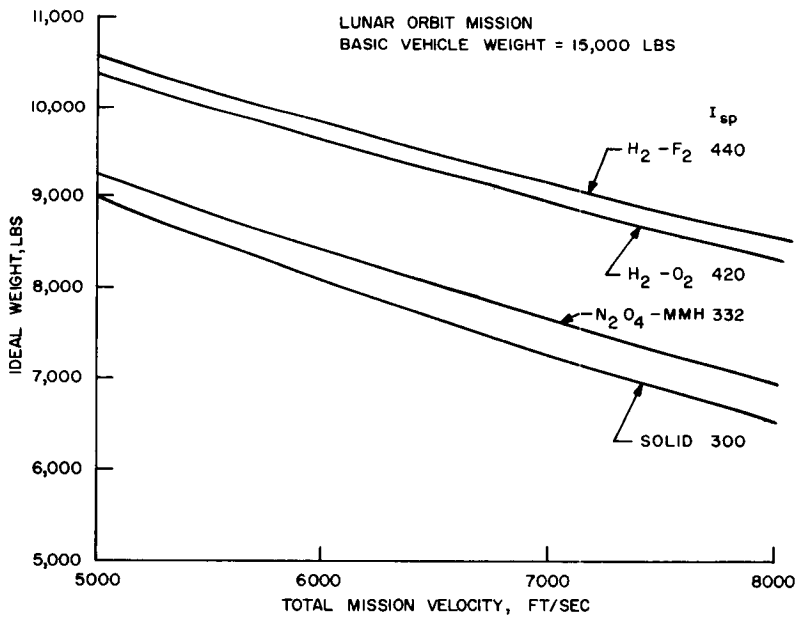


Figure XIII-3-2. Ideal weight versus mission velocity, lunar orbit mission

CONFIDENTIAL

Studies have also been conducted to determine the weights of pressurized versus pumped systems. A summary of such data appears in Table XIII-3-4, for the high-energy cryogenic propellants. It is obvious that the pumped system is lighter when both pressurized and pumped units are operated at chamber pressures of 300 psia. However, if the chamber pressure is reduced to less than 100 psia for the pressurized system the weight difference is significantly reduced. Operation at 60 psia will probably eliminate any weight advantage with the type of pressurization system utilized herein. Since the lower chamber pressure is contemplated for the pressurized system, the use of a turbo-pump will only add an additional source of complication.

TABLE XIII-3-1. PROPULSION DATA — CONSERVATIVE SYSTEMS

CONDITIONS:

Take-off Weight	15,000 pounds
Abort Propulsion System Weight	850 pounds
Vehicle Weight at Escape	14,150 pounds
Velocity Requirement (mid-course plus lunar orbit and orbit escape), modified ellipse trajectory	7300 ft/sec

		Solid	Storable N_2O_4-MMH	Cyrogenic $H_2 - O_2$
1.	Weight at Escape	14,150	14,150	14,150
2.	Chamber Pressure, psia	—	300	300
3.	Expansion Ratio	25	25	25
4.	Specific Impulse, sec.	282	306	410
5.	Mid-course Propulsion Weight	690	560	425
6.	Weight at Entry into Lunar Orbit	9350	9900	10,850
7.	Weight at Escape from Lunar Orbit	6150	6600	7050
8.	Allowable Vehicle Weight	5460 (6200)*	6100 (6650)	6625 (7200)

* Numbers in parenthesis are for a typical figure 8 trajectory having a mid-course ΔV of 500 feet per second and an orbit and orbit escape requirement of 6100 feet second. The total required ΔV is thus 6600 feet per second.

TABLE XIII-3-2. PROPULSION DATA — 1966 SYSTEMS

CONDITIONS:

Take-off Weight	15,000 pounds
Abort Propulsion System Weight	600 pounds
Vehicle Weight at Escape	14,400 pounds
Velocity Requirement (mid-course plus lunar orbit and orbit escape), modified ellipse trajectory	7300 ft/sec

		N_2O_4-MMH	$H_2 - O_2$	$H_2 - F_2$
1.	Weight at Escape	14,400	14,400	14,400
2.	Chamber pressure, psia	150	60	60
3.	Expansion Ratio	40	40	40
4.	Specific-Impulse, sec.	322	427	440
5.	Propulsion System Weight	7370	6130	5910
6.	Allowable Vehicle Weight	6650 (7140)*	7380 (7870)	7790

* Numbers in parenthesis ( ) are for a typical "figure-8" trajectory having a mid-course ΔV of 500 feet per second and an orbit and orbit escape requirement of 6100 feet per second. The total required ΔV is thus 6600 feet per second.

TABLE XIII-3-3. WEIGHT DATA FOR PRESSURIZED SYSTEMS

ΔV = 7300 ft/sec

PROPELLANT COMBINATION	O_2/H_2		F_2/H_2	
THURST/WEIGHT RATIO	1.5	2.0	1.5	2.0
Chamber Pressure, PSIA	100	100	100	100
Expansion Ratio, E	40	40	40	40
Oxidizer Weight, lbs	5675	5675	6177	6177
Fuel Weight, lbs	1425	1425	428	428
Tank & Pressurization Weight, lbs	870	870	310	310
Thrust Chamber Weight, lbs	895	1360	1255	1795
Total System Weight, lbs	8865	9330	8170	8170

TABLE XIII-3-4. COMPARISON OF PRESSURIZED AND PUMPED SYSTEM WEIGHTS

TOTAL VEHICLE WEIGHT = 15,00 pounds

ΔV = 7300

	TOTAL PROPULSION SYSTEM WEIGHT, LBS	
	O_2/H_2	F_2/H_2
Pumped - Pc = 300 psia	7750	7330
Pressurized - Pc = 300 psia	9000	8500
Pressurized - Pc = 100 psia	7900	7500

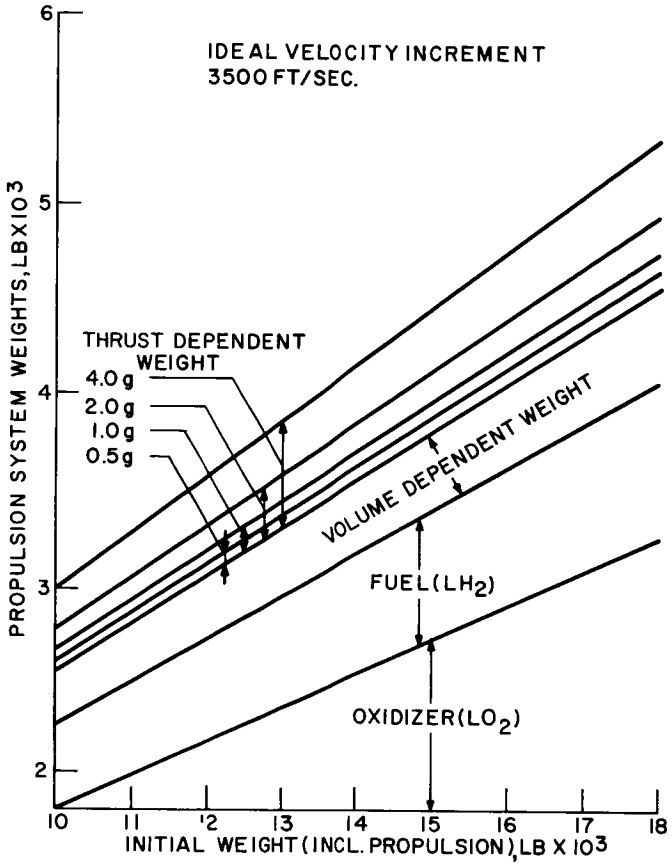


Figure XIII-3-3. Propulsion system weights

CIRCUMLUNAR MISSION

This data is presented to illustrate the propulsion requirements if no lunar orbit were required. Since the circumlunar mission requires considerably less energy than a lunar orbit, several changes in assumptions have been made. The first assumption is that the entire APOLLO vehicle is to be saved in an abort situation. An analysis of the propulsion weight for solid propellant units indicates an abort rocket weight of approximately 1550 pounds. Because of the high-thrust — short-duration nature of the abort system, it is not likely that it can be used for mid-course correction. It is thus assumed that the abort system will be dropped at or prior to achieving escape velocity. The weight of the vehicle at escape is, conservatively, 15,000 minus 1550 pounds or 13,450 pounds.

A total mid-course velocity correction capability of 600 feet per second has been used for this study, split between the outbound and return portion of the flight.

Utilizing methods similar to those for the lunar orbit mission, the data in Table XIII-3-5 was prepared for three propellant types. The allowable vehicle weights appear ample for all of the propulsion systems. Only a small weight advantage accrues from using liquid oxygen and liquid hydrogen compared to the storable propellant combination. Additional study will be necessary to choose between the storable liquid and the solid propellant systems. The liquid system offers hypergolic ignition, ease of control, and a small number of units. The solid propellant has inherent simplicity at the expense of requiring a large number of units to obtain a degree of control flexibility.

TABLE XIII-3-5. CIRCUMLUNAR MISSION SUMMARY DATA

	Solid	N_2O_4 - MMH	H_2 - O_2
1. Total Vehicle Weight	15,000	15,000	15,000
2. Abort System Weight	1550	1550	1550
3. Weight at Escape Velocity	13,450	13,450	13,450
4. Mid-course Propulsion Weight	1800	960	770
5. Allowable Vehicle Weight	11,650	12,490	12,680

LUNAR LANDING MISSION

A limited amount of effort has been expended to determine first-pass approximations for a vehicle capable of a lunar landing and take-off. Data is included for liquid-oxygen —liquid-hydrogen, and liquid-florine — liquid-hydrogen combustions. It has been assumed that staging occurs between the lunar landing and take-off, leaving the expended hardware behind. Velocity requirements of 250 feet per second for outbound mid-course correction and 8500 feet per second for a zero-velocity impact on the Moon's surface were used. Identical values were used on the return flight, requiring 8750 feet per second for each portion of the flight. Assuming a return vehicle weight of 8850 pounds results in an APOLLO vehicle weight at launch equal to 39,000 pounds for hydrogen-fluorine and 47,000 pounds for hydrogen-oxygen. Utilizing standard methods of analysis, the data in Table XIII-3-6 was prepared.

TABLE XIII-3-6. LUNAR LANDING - PRELIMINARY DATA

Total Mission Velocity = 17,500 ft/sec

		$H_2 - O_2$	$H_2 - F_2$
1.	Specific Impulse, sec.	430	455
2.	Expansion Ratio	60	60
3.	Tankage Wt. and Pressurization Wt. per cubic foot of propellant, lb.	3.0	3.0
4.	Weight at Escape, lb.	47,000	39,000
5.	Weight at Lunar Landing, lb.	25,400	21,400
6.	Weight at Lunar Take-off, lb.	20,700	18,600
7.	Return Payload Weight, lb.	8850	8850

4. RELIABILITY

The reliability aspects of the propulsion system for manned space flight cannot be over-emphasized. The abort system is the sole mechanism for escape in case of booster malfunction. For other non-catastrophic failures, it represents the means of shortening the mission to effect a successful re-entry. Thus, basic required reliability becomes the solid ground on which the initial preliminary design must be built.

Long a primary objective at MSVD, reliability must be established as an early design parameter, built into the components developed, demonstrated, and maintained throughout the development program. The reliability program may be logically divided into four phases. These are as follows:

A. Design (Reliability Synthesis)

B. Development Program (Reliability Attainment)

C. Demonstration Test Program (Reliability Measurement)

D. Production and Quality Assurance (Reliability Maintenance)

During the design, reliability attainment is achieved through careful attention to both expected performance and safety. A series of analyses will be made to analyze the complete spectrum of any potential malfunctions throughout the flight period. These malfunction analyses then permit trade-off studies to be performed to relate potential improvements in reliability to mission parameters such as weight, cost, etc.

An example of reliability attainment is through the use of redundant motors versus unit reliability. A typical example is shown in Figure XIII-4-1. This reliability gain must then be weighed against the obvious losses of decreased payload weight, complexity and cost. During the present study, effort will be made to delineate those trade-off studies necessary for preliminary design which may appreciably affect reliability. Where time permits, an example or two illustrating these trade-off studies will be included in the study.

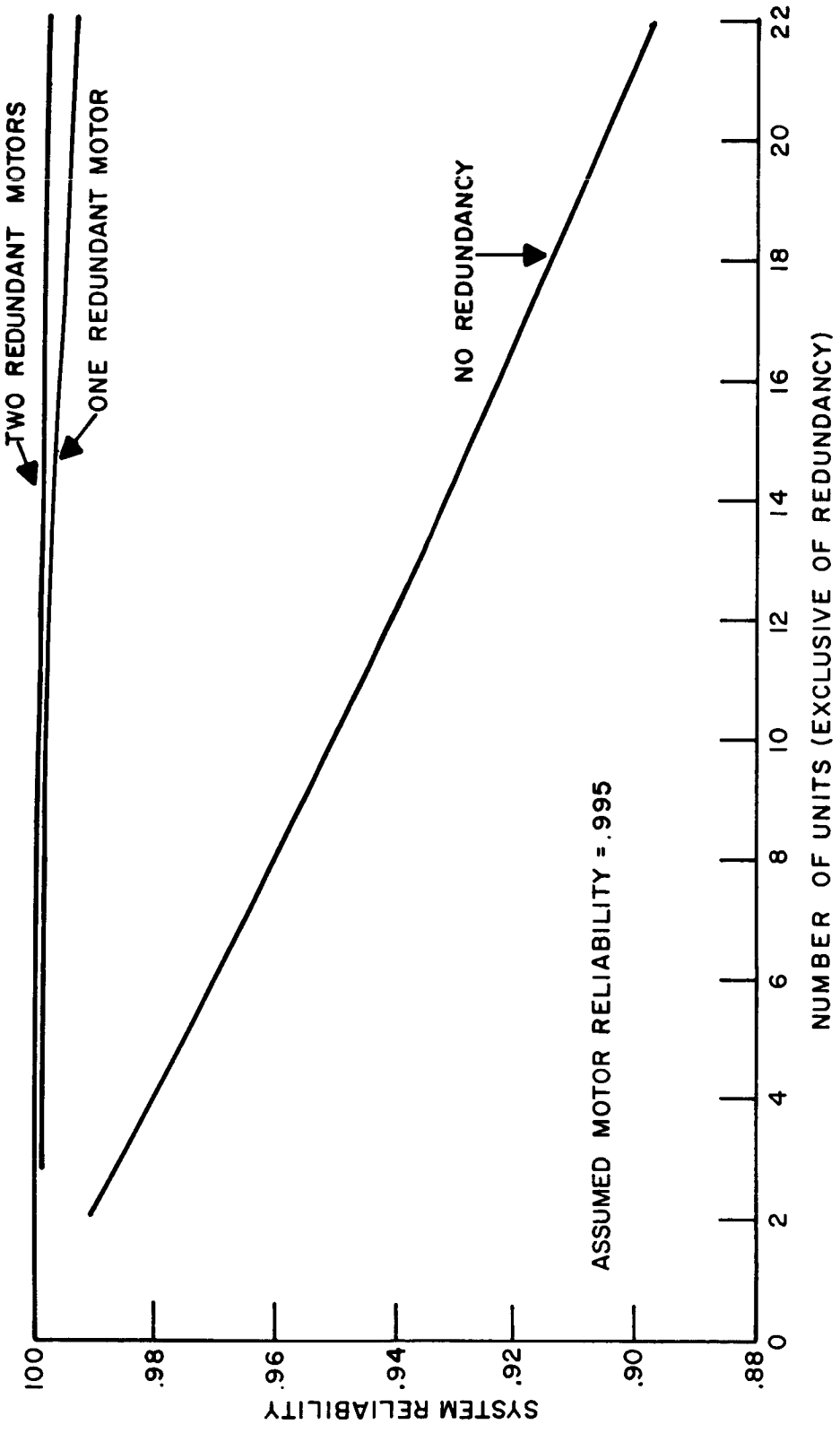


Figure XIII-4-1. Typical propulsion system reliability

For the advanced propulsion needs of APOLLO, it is mandatory that plans include ample consideration of reliability in the development program. With limited time and funds, the balance of work between design and development must be maintained to assure an adequate and safe propulsion system which will reflect the "state-of-the-art" at the time of completion of development testing.

Emphasis on adequate development of reliability rather than demonstration is the key here. It does little good to demonstrate solid abort rocket propellants with conventional steel cases if the mission requires a higher strength to weight case - such as the fibre glass wrapped cases under development at Aerojet-General. Obviously development of these cases must attend the development of the required higher performance propellant to assure an adequate, safe abort rocket system.

What then of reliability demonstration? The answer here is an intelligent balance between demonstrated reliability, confidence level and program cost. The development and demonstration programs need to be designed to provide a best use of a modest budget, thereby best overall probability of mission success. For example, it might be advantageous to continue development of reliability of an abort rocket from, say, 0.98 to 0.995, even though only enough reliability demonstration tests can be conducted to produce a confidence level of 50 percent in the higher reliability. This is in contrast to alternate programs which might either spend the same amount of money running repeated tests of existing units to demonstrate the reliability of 0.98 with a confidence level of 97.5 per cent; or an alternate design program to increase unit redundancy to give an overall 0.955 reliability, but at the expense of increased weight and complexity.

Rigorous discipline and analysis are necessary here to insure proper attainment of reliability, safety and optimum utilization of propulsion technology.

The final area (D, above) of reliability maintenance remains immutably fixed as the most necessary adjunct to safe and reliable propulsion. None of the foregoing analyses can compensate in any manner for adequate production and quality control. Production of a reliable solid abort motor, for instance, can only be achieved with careful attention to each step of the fabrication. All appropriate quality control techniques such as grain x-ray, case zyglo, magnaflux, dimensional inspection, batch quality

control, etc. are necessary for development, proof test and production of propulsion systems. When defects occur, for example, the separation of the solid charge bonding to the wall, it is possible to analyze these defects during development test and develop an adequate knowledge of the nature of limitations of such components and a quality assurance code for final production units.

Such a program of continued reliability analysis, attainment and maintenance, rather than strict adherence to any existing MIL specifications for manned rocket engine, should ultimately produce the most reliable and safe propulsion system for the manned APOLLO mission.

An attractive oxidizer, OF_2, appears to have some merit for advanced versions of space vehicles. Reaction Motors Division of Thiokol has produced and studied small quantities and is planning tests on small thrust chambers in the near future. Vehicle payloads with OF_2 and MMH should exceed those with H_2/O_2 being approximately equivalent to those with H_2/F_2. Properties and storability of OF_2 need further study, as well as measured performance. Such a combination as OF_2/MMH would permit storage at warmer temperatures than the hydrogen/oxygen system. Studies of OF_2/MMH will continue for an advanced system.

5. FUTURE EFFORTS

Presented below is a brief outline of further studies that are being conducted to assist in the selection of the on-board propulsion system.

A. Individual System Optimization
   1. Obtain operating parameters
   2. Determine Weights and Volumes

B. Provide data for trade-off studies
   1. Establish thrust levels
   2. Establish on-off requirements

C. Select Integrated System

D. Prepare Preliminary Design
   1. Establish Reliability level
   2. Investigate system redundancy

E. Miscellaneous Study Areas
   1. Propellant Acquisition under zero "g" conditions
   2. Investigate cryogenic problems
   3. Design Attitude Control System
   4. Determine system arrangement

F. Propulsion Energy Management Analysis

The major portion of these studies is being performed by Aerojet, Thiokol, and Bell. These companies have placed experienced and competent personnel on this project and will be able to draw on their large backlog of experience in liquid and solid propulsion. The General Electric Company has provided input data to these organizations and will monitor and integrate the efforts of the previously mentioned organizations. The skills and techniques that these groups bring to bear will enhance the propulsion aspects of the APOLLO Study.

6. PROPULSION SYSTEM CONSIDERATIONS

PROPULSION SYSTEM SELECTION

This will be largely dictated by the mission requirements for a highly reliable, manned rocket engine capable of repeated operation during the 14-day mission. Specific design analyses are in progress to define the actual propulsion components to meet the necessary velocity increments. Toward this goal, considerable analysis and preliminary design work is underway at Thiokol Chemical Corporation and Aerojet Solid Rocket Plant to define solid propellant systems; and at Reaction Motors Division, Aerojet Liquid Rocket Plant, and Bell Aerosystems on Liquid propellant systems. Propulsion systems are being integrated into the APOLLO vehicle with system engineering management by General Electric Missile and Space Vehicle Department.

Several general observations have emerged from the work to date which permit definition of the propulsion system.

1. The high thrust, short burn-time, and installation requirements dictate that the abort propulsion and separation rockets should be solid propellant. There are a number of suitable solid propellant rockets available which well suit the APOLLO requirements; a typical unit is shown in Figure XIII-6-1. This motor has been developed by the Elkton Division of Thiokol Chemical Corporation.
2. The incremental velocities for orbit and orbit excape at the Moon dictate that high energy propellants are necessary. This indicates liquid hydrogen with liquid oxygen or liquid fluorine are essential for this mission based on today's propulsion technology; although other high energy systems should be considered for future development.
3. Assuming cryogenic liquid propellants, with light-weight propellant pressurization systems, the heat balance for the 14-day mission indicates substantial heating of residual propellants. This suggests that the tank should be of sufficient strength to withstand moderate pressure (in the vicinity of 100 psia for liquid hydrogen) and capable of completing the 14-day mission without

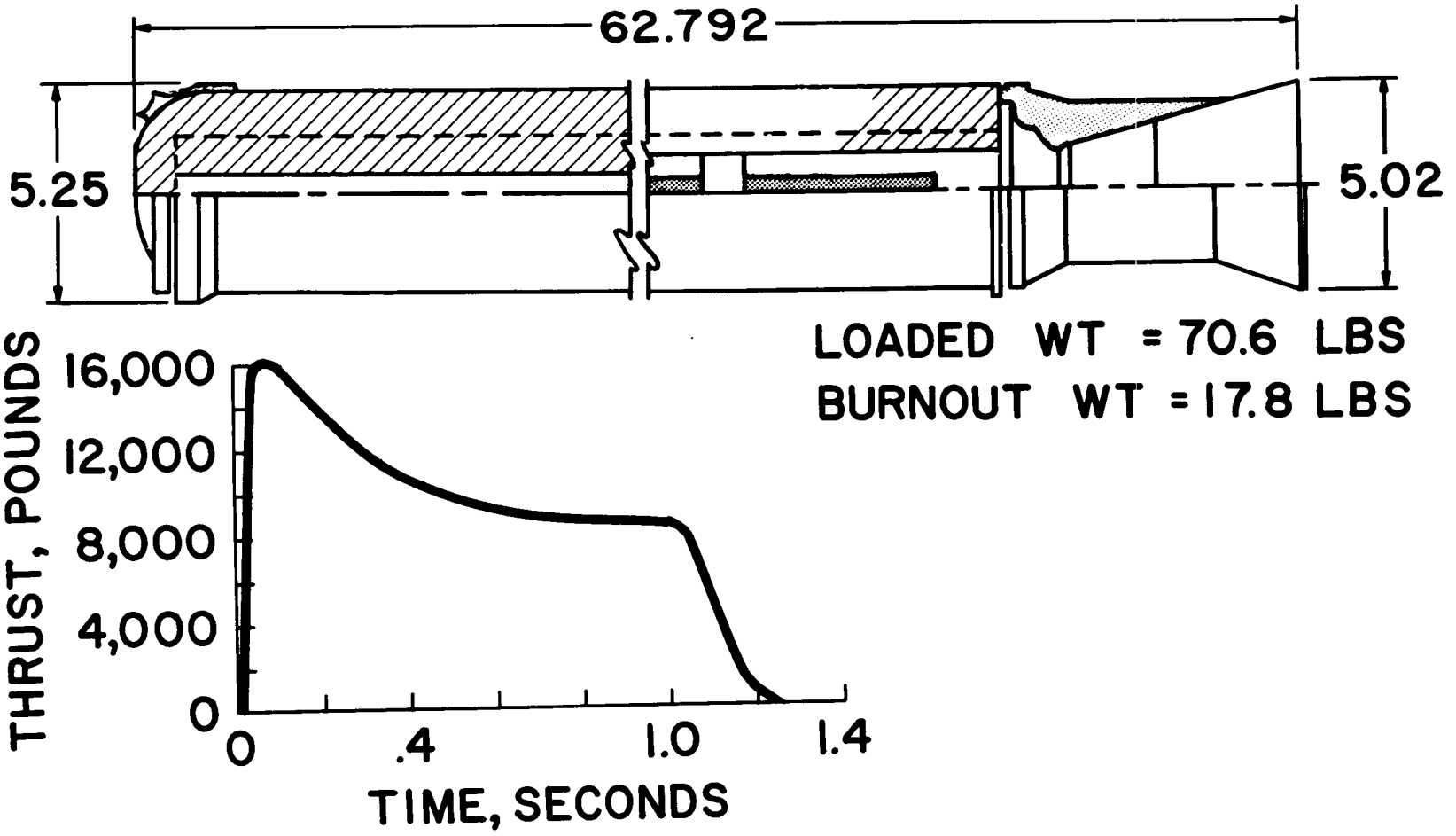


Figure XIII-6-1. Thiokol Chemical Corporation TE-146

venting. (Venting to release pressure from heating is both wasteful and difficult in this mission.)

4. If room permits, the main rocket propellants should be supplied from pressurized tanks - rather than by pumps. Only about 1 percent loss in performance of hydrogen with oxygen or fluorine is experienced at chamber pressures as low as 40 to 60 psia. If tanks need to withstand approximately 100 psia for the two-week mission, there is little need for pumps. Further, with pressurized systems there are several significant advantages for this mission:

   a. Starting of engines is simplified, without protracted cool-down periods. Engines can be started at low flow regardless of whether flow is liquid or vapor, thereby achieving positive acquisition of propellants.

   b. The low-chamber pressures permit use of uncooled ablative chambers, thereby eliminating the coolant pressure drop, cooling unreliability, propellant loss during start and shutdown, and permitting a more precise total impulse control during engine start and shutdown.

   c. Small, precise increments of velocity can be achieved with just the heated hydrogen. (Specific impulse of H_2 gas is 200 seconds at 270° R.)

   d. Toward the end of the mission, it is anticipated that ever-warmer propellants will be burned, winding up with combustion of all vapor at the end of the mission for most effective propellant utilization.

Where space for propulsion is limited, reduction in required volume can be achieved as discussed below. The largest volume is required for the pressurized H_2/O_2 system. Volume can be reduced by going to a pumped system to reduce thrust chamber size, or to H_2/F_2 to reduce tank size, or to a pumped H_2/F_2 system where volume is severely restricted.

Available volume of the D-2 configuration permits use of a H_2/O_2 pressurized system with its attendant higher reliability and facilitated storage. The glide vehicle propulsion package volume appears limited, and, as such, a pumped H_2/F_2 system is shown. Advantages of fluorine, in addition to lower volume requirements, are hypergolicity and slightly improved specific impulse. Disadvantages of fluorine center

around the special handling and use precautions resulting from the extremely high chemical reactivity and toxicity of F_2 and its reaction products.

PROPELLANT TANKAGE

Perhaps the key to successful utilization of cryogenic, high energy propellants is the successful storage and expulsion during the 14 day mission. Heat leaking into the propellants must be minimized by minimizing tank surface area, use of good insulation such as Linde SI-4 plus utilization of the vacuum of space, suitable tank supports, and an adequate pressurization system. Venting of the propellant tanks, to relieve the pressure built up by this heat is difficult to achieve for this mission and wasteful of propellant energy. Proper design and insulation of tanks should permit a total pressure build-up of no more than 100 psi. Since minimum gage problems dictate that tank walls will stand at least 100 psi, we do not plan to vent the cryogenic propellant tanks throughout the mission.

The available space permits storage of the hydrogen in a single sphere. This provides minimum tank weight for this largest volume tank, and minimum surface area to insulate. The liquid oxygen may be stored in a torus or sausage tank as shown, or in individual spheres.

Pressurization may be accomplished by a combined system which gives a belt and suspenders redundancy for safety. The hydrogen and oxygen are each pressurized with the corresponding vaporized propellants, the hydrogen liquid by hydrogen gas at 150°F, the liquid oxygen by oxygen gas at 360°R. This is similar to the Aerojet Hylas pressurization system, demonstrated during 40 tests under AF contract AF 04(611)-5170, but with heat supplied for vaporization by combustion of $H_2/0_2$. The second system utilizes the energy contained in the propellant to generate vapor pressure for self expulsion. This second method appears most feasible during the latter part of the mission when there is a large volume of gas over the liquid. In addition, the liquid temperature has risen from heat leaking in. During early phases, this self pressurization system can be employed with some heating of propellants to maintain vapor pressurization at an adequate value.

A detailed analysis of this pressurization system is underway now by General Electric and the propulsion sub-contractors.

D-2 MAIN ROCKET ENGINE DISCUSSION

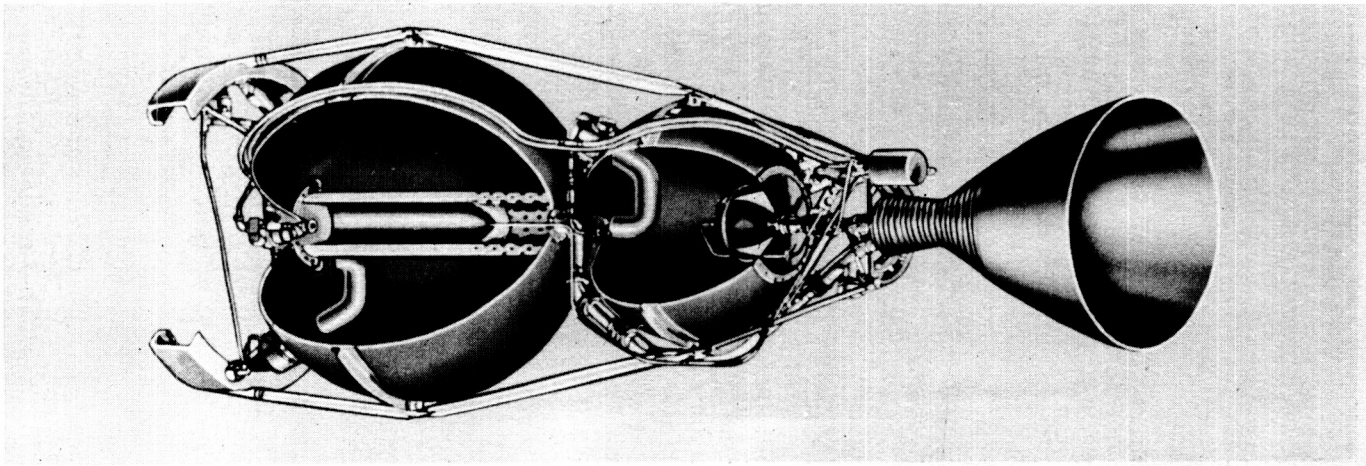
The propulsion package for the D-2 configuration will utilize existing technology and components, where suitable, to provide a simple, reliable, relative high performance rocket engine system. Selection of a pressurized propellant fed system facilitates achievement of these goals through use of simple, uncooled ablative thrust chambers, similar to those developed by Aerojet General under Contract AF 04(611)-5170. Aerojet's Hylas engine, which evolved from this work, is shown on Figure XIII-6-2.

For this discussion, the engines shown in the D-2 configuration are similar to the Hylas engine, although there are several other propulsion contractors who have equivalent engines which will be considered before final selection of a propulsion system. Other equivalent engines include the Rocketdyne NOMAD, the de-rated Bell HYFUS and the de-rated Pratt and Whitney CENTAUR. These, however, would in each case require some modification or development for the APOLLO vehicle.

At our design chamber pressure of 60 psia, the Aerojet ablative chambers have been demonstrated during tests of 300 seconds without measurable change in throat area. The use of uncooled chambers, provides the inherent reliability of simplicity, and eliminates the necessary cooling pressure drop, time delay, and loss of residual propellants in the cooling passages. For redundancy, four chambers each independently operable will be used. Each chamber, with a thrust of approximately 6000 lbs., will be capable of performing the complete mission. This provides a substantially improved system reliability. All four chambers can be fired in an emergency - such as a super-orbital abort - if the requirement should exist.

The chamber injectors will be similar to those developed under the Aerojet Hydra program (Contract AF 04(611)-5170) which demonstrated a vacuum corrected specific impulse of 433 seconds at 40 psia and expansion ratio of 40. During these tests, the total pressure drop from the tanks to the chamber was less than 15 psia.

CONFIDENTIAL



GROSS WEIGHT, LB	5,000
THRUST, LB	6,000
CHAMBER PRESSURE, PSIA	40
SPECIFIC IMPULSE, SEC	433
TOTAL IMPULSE, LB-SEC	1.80×10^6

CONFIDENTIAL

Figure XIII-6-2. Aerojet-General Corporation "Hylas"

The chambers are gimbaled through an angle of approximately five degrees to assure alignment of thrust through the vehicle center of gravity. Control of the chambers will be furnished by the attitude control sensing system. The chambers need not swing rapidly since vehicle attitude will be maintained by the separate attitude control system.

Space restart capability is incorporated in this system through utilization of inherent advantages of the pressurized feed system plus electrical ignition. The pressurized propellants are supplied to the chamber in a low-flow start sequence for ignition and propellant acquisition during the resultant low thrust. The initial flow may consist of either liquid or vapor without detrimental effects (unlike pumps which require a cool-down period and assurance of liquid fuel and oxidizer flow). Ignition is assured either through separate igniter chambers such as in Thiokol's XLR99RM-1, or separate high energy spark plugs directly in the injector face. Use of a spark plug like the surface gap spark plug assures ignition even under the low pressures during start transients.

Propellant valves are located near the injector inlets and operated together for reliability. Additional safety valves are utilized in the lines and at the tank outlets for safety. The low pressures permit selection of available flex lines to the gimbaled chambers.

The rocket engines so far discussed are based on existing technology and designs, and require only development in the state-of-the-art. The coupling of the low pressure feed system with the low (60 psia) chamber pressure ablative cooled chambers provides an inherently simple, safe, low-cost system with early availability, yet without significant sacrifice in the high performance objectives. The design is both conservative and sound, based on actual test data of chambers and the proposed pressurization system, and can be developed immediately using existing technology, materials, and techniques.

ADVANCED (REVERSE-FLOW) ROCKET ENGINE DISCUSSION

There are a number of advantages to be garnered from an integration of the thrust chamber and tankage with the space vehicle structure. One method of achieving this integration is through use of the reverse-flow nozzles now being studied by several

propulsion contractors, including the General Electric Rocket Engine Section at Malta Test Station under contract AF 04(611)-6016. One configuration using a reverse-flow nozzle called an "inverted plug" is shown in Figure XIII-6-3. There are also other variations which will be examined.

Basically, the reverse flow nozzle produces an extremely short chamber by expanding the supersonic flow as a corner expansion around the lower chamber lip to exit axially. Segmenting the chamber into 8 units permits adequate redundancy to assure mission safety. Since the chamber is constructed from uncooled, ablative elements, the outer walls can double-in-brass for the outer vehicle structure, as well as transmit the loads directly to the propellant tanks, mission module and command module.

Since the thrust chamber can now take advantage of the complete available vehicle diameter for rocket gas expansion, the chamber expansion ratio can be raised from 40 to approximately 200, still retaining the low chamber pressure of 60 psia. This should result in an increase in specific impulse for the H_2/O_2 system from 427 seconds to 450 seconds, or for the H_2/F_2 system from 440 to 460 seconds.

The principal gain here is in better packaging of the space vehicle, with an attendant reduction in vehicle length and possibly weight. Studies will be continued to evaluate potential gain of advanced chambers in more detail.

ALTERNATE PROPELLANT CONSIDERATIONS

There are several other fuels which can be considered for this application if hydrogen proves disadvantageous. These include the amine fuels, such as MMH, UDMH, hydrazine, and even ammonia. In each case the specific impulse would be reduced, however, and for a low-pressure system such as that proposed, the increased density pays only modest dividends. Boron, lithium, and beryllium fuels have yet to prove their worth. On the other hand, there is a rapidly increasing wealth of data on hydrogen, all of which indicates its excellence as a rocket fuel, if adequate handling and storage procedures are employed.

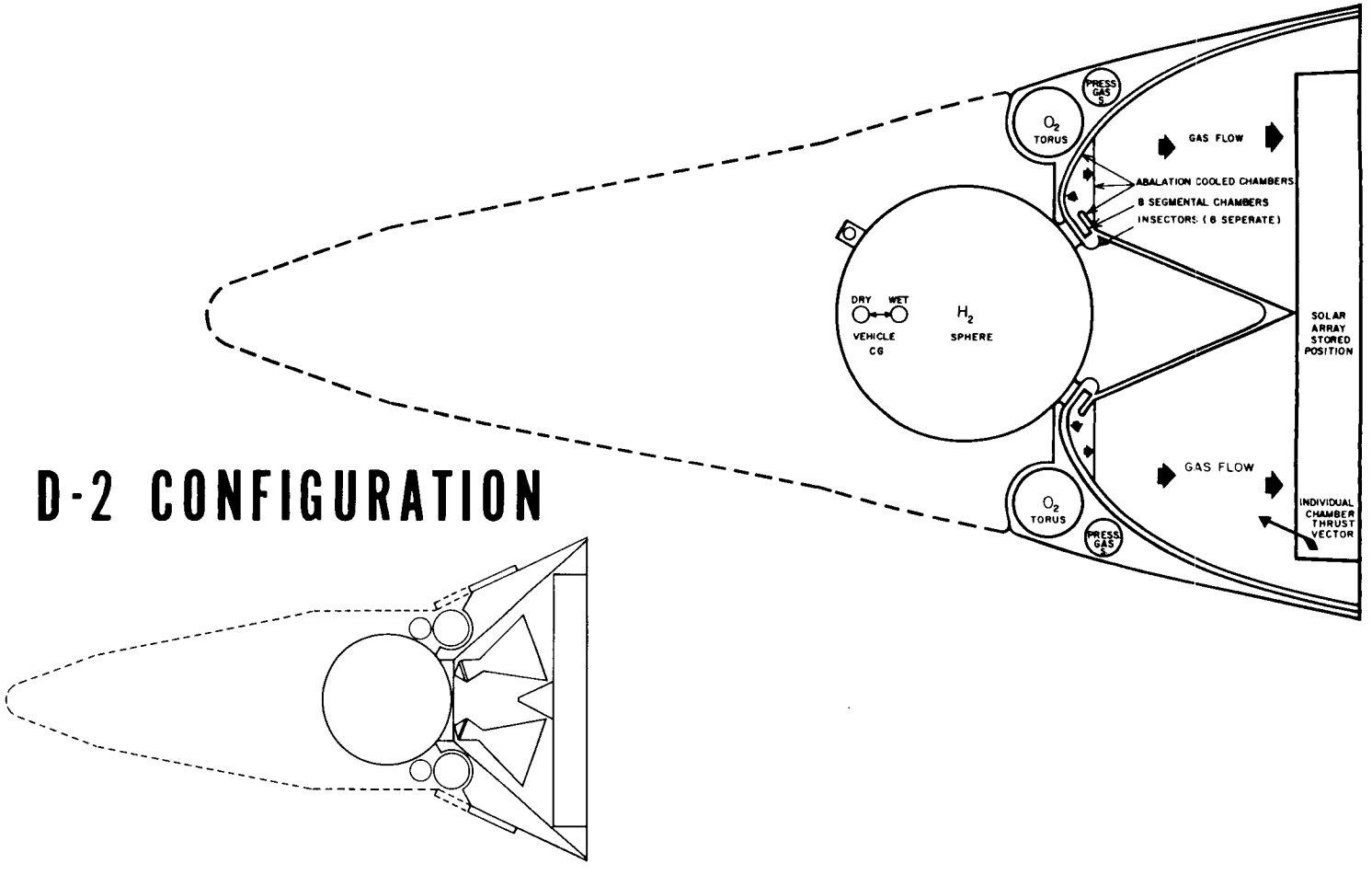


Figure XIII-6-3. Proposed advanced propulsion system

CONFIDENTIAL

CHAPTER XIV

GROUND SYSTEMS

XIV. GROUND SYSTEMS

1. GENERAL

The ground support equipment in the ground system of the APOLLO Project will be required for the period from delivery of the APOLLO launch vehicle from the factory to delivery of the recovered re-entry vehicle to its final location — either at General Electric's Missile and Space Vehicle Department (GE-MSVD), or some other final location designated by NASA. Listed below are the major areas where ground support equipment will be located and the types of equipment that will be included.

FACTORY AREA

One of the dual functions of this equipment is to supplement that equipment to be provided by Quality Control and Test Operation (QC&T) for in-house component, subsystem, and system acceptance tests of the APOLLO launch vehicle. Since similar equipment must be developed and provided by QC&T for their in-house use, all or any part of the equipment listed below will be made available to them. Such multiple use will eliminate the duplication of development costs on like equipment. In any event, the equipment included herein will be used for system and subsystem evaluation and compatibility tests.

1. Handling Equipment.
2. Checkout Equipment — quantitative and qualitative, electrical, electro-mechanical, pneumatic, and/or hydraulic.
3. Servicing Equipment — liquid oxygen, liquid nitrogen, liquid propellant, cooling air, etc.
4. Ground Instrumentation Equipment.
5. Shipping Equipment.

FIELD HANGAR AREA

1. Handling Equipment.
2. Checkout Equipment — same as in "Factory Area."
3. Static Balance Equipment.
4. Servicing Equipment — same as in "Factory Area."
5. Ground Instrumentation Equipment.

HANGAR– LAUNCH PAD TRANSPORTING EQUIPMENT

EXPLOSION INSTALLATION AREA

1. Handling Equipment.
2. Checkout Equipment.

FIELD LAUNCH AREA

1. Handling and Mating Equipment.
2. Confidence Checkout Equipment.
3. Control and Monitoring Equipment.
4. Servicing Equipment.
5. Ground Instrumentation Equipment.

SEARCH AND RECOVERY AREAS

1. Search Equipment Located on Land Vehicles, Aircraft, and/or Surface Vessels.
2. Recovery Equipment Located on Land Vehicles, Aircraft, and/or Surface Vessels.

GROUND TRACKING AND COMMUNICATION AREAS

1. Special Ground Tracking Equipment required for the APOLLO Vehicle.
2. Special Ground Communication Equipment required for the APOLLO Vehicle.

2. OPERATIONAL SEQUENCE

The following discussion, an amplification of a portion of the Mission Profile (Chapter II), presents a normal operational sequence in the handling and checkout of the APOLLO Vehicle for all operations from the factory through recovery of the APOLLO Re-entry Vehicle and its crew and, finally, delivery to their ultimate destination after recovery. Although there are a number of different vehicle configurations under consideration, this description applies to all of them; however, where one of the configurations requires a deviation from the operational sequence described below, this exception will be specifically called out.

FACTORY AREA

Upon the completion of the factory subsystem and system acceptance tests of the APOLLO launch vehicle by the Quality Control & Test Operation, the vehicle will be prepared for shipment to the launch complex at Cape Canaveral, Florida. For shipping purposes only, the booster adapter section (or main propulsion module) will be disassembled from the launch vehicle. Because of the large diameter of the booster adapter section, it will be made in halves to make for ease of shipment to the field. The exact method of shipment has not been determined as yet. However, the contemplated sizes of the shipping configurations prevent the use of rail shipment. Therefore, studies on various methods of shipping the disassembled vehicle by air, highway, and, possibly, water will be continued. All rockets, explosive, and pyrotechnic devices will be shipped separately.

FIELD HANGAR AREA

When the APOLLO space vehicle, in its three basic shipping subassemblies, arrives at Cape Canaveral, the two halves of the booster adapter section will be assembled together and mounted on a combination handling fixture and stand. The third shipping subassembly — the remainder of the APOLLO space vehicle — will be mounted on the transport vehicle that will be used for transporting the fully assembled space vehicle

within the hangar area itself and between the hangar area and launch pad. A space vehicle handling fixture, assembled to the space vehicle at the factory, will be used in removing the APOLLO space vehicle subassembly (less booster adapter section) from its shipping configuration and assembling it to the transport vehicle. It is conceivable, but not definite at this time, that the APOLLO space vehicle subassembly in its handling fixture will be assembled to the transport vehicle at the factory, and this combination will be shipped as a single unit to the field.

At this time, the booster adapter section, assembled together in its own handling fixture, will be assembled to the space vehicle subassembly mounted on the transport trailer. The APOLLO space vehicle is now fully assembled and ready for a complete hangar checkout as described below.

1. Visual Inspection

A complete visual checkout of the space vehicle will be made to insure that no damage occurred due to shipment from the factory to the hangar area.

2. Subsystem Checkout

Using a space vehicle checkout console, a complete electrical checkout, utilizing a ground power supply, will be made on each vehicle subsystem. The checkout console will control, monitor, and perform all of the detailed tests to isolate any faults in the subsystem down to the black-box component level. Both an electrical ground disconnect and an inflight disconnect will be used for electrically connecting the checkout console to the space vehicle.

A complete quantitative test of the electronic subsystem will be performed utilizing an electronic subsystem test set. This test of the instrumentation and communications Subsystem will include such tests as:

a. Check of receipt and execution of commands.

b. Verification of beacon response.

c. Confirm proper operation of voice links, optical tracking aids, and telemetry channels.

The electronic subsystem test set will be housed in an air-conditioned mobile van that can be used at the explosive installation area, and the launch pad, as well as in the hangar area. This equipment will be capable of testing the electronic subsystem by either hardwire through the electrical ground disconnect or by free radiation.

By means of a vehicle dynamic test machine, a complete dynamic test of the guidance and control subsystem will be made. Using an automatic programmer controlled through a dynamic test console, the guidance and control subsystem will be checked for correct response in each of the three axes of roll, pitch, and yaw. Correlation of the telemetry response will also be obtained at this time. Where possible, simulation of all one-shot type of devices — such as squibbs, rockets, etc. will be provided.

A complete check will be made of the environmental subsystem using the vehicle checkout console. These checks will include verification and calibration of the associated instrumentation and crew displays. Such parameters as partial pressures of oxygen, nitrogen, carbon dioxide, and water vapor, as well as cabin temperature and pressure will be measured and compared with the readings from the displays for verification of calibration.

3. Plumbing System Leak Detector Tests

After the vehicle system has been successfully checked, leak tests will be made of the vehicle plumbing systems — oxygen, nitrogen, pneumatic, and/or hydraulic. These tests will be similar, in nature, to the checks that have been made in the past on other space vehicle and re-entry vehicle systems developed by GE-MSVD. The tests will be run with the tanks pressurized to the "on stand" pressure. The pressure in each system will then be monitored for a sufficient length of time to verify that the system is tight.

In the event that leaks are detected, a small amount of helium will be used to isolate the leak. During these tests, certain valves will have to be energized on command from the vehicle checkout console, and may, therefore, be disconnected from the airborne commands and tied back into the ground support equipment.

For this reason, this test must precede the full system checkout, since the latter will verify that these solenoids have been reconnected to the airborne system. If high

pressures and large volumes of gases are involved in any of these systems, these tests will be performed in a special high-pressure test area. The operating personnel will be protected against a catastrophic failure of the system.

4. Cabin Area Leak Detector Tests

A leak detector test is required to determine the airtight integrity of the cabin area, as well as the bulkhead door between the mission module, and the reentry vehicle module. A helium tracer gas leak detector, developed on a previous bio-medical program by GE-MSVD, will be used for these tests. This equipment utilizes the mass spectrometer principle, and the basic design can be adapted for the APOLLO vehicle. This particular system will not be affected by the heat generated due to the occupants or the operation of any electronic equipment. This will be a quantitative type of test, insuring that the vehicle will not incur a total leakage in excess of the maximum allowable. If the cabin area has a total leakage which is more than is tolerable, suspected sources of leakage will be subjected to a jet of helium to pin point the location of the leak.

5. Static Weight and Balance Checks

Analysis of re-entry vehicle dynamics in this and other re-entry vehicle programs has shown that a balance check in the field, in addition to the complete balancing procedure performed in the factory, is essential to assure impact of the re-entry vehicle within the specified impact area and to minimize the control forces required. A vehicle leaving the factory would be dynamically balanced, weighed, and the moments of inertia and the center of gravity measured.

While in the field, such operations as retrofitting, replacing of dummy with actual components and variations of crew weight will affect the balance. As each change to the balance is performed, a notation describing the weight and location of the change will be recorded in a log book. Following the last modification, several calculations will be performed to determine the respective changes to the balance, weight, moment of inertia, and center of gravity location.

Repeating the complete factory balancing operation in the field to check the calculations would be impractical and unnecessary. However, an actual check to the weight and center of gravity location is possible and necessary, and a significant difference

between the actual and calculated values of the weight and center of gravity would point to faulty calculations of the balancing characteristics. For measuring the weight and center of gravity of the APOLLO space vehicle in the field, a weight and static balancing machine will be used which is rugged and operationally simple, yet capable of making the necessary measurements with the actual crew and complete flight equipment in place. A weight and balance machine, operating on a force beam principle, has been developed on a past program with successful results, and it is certain that this type of machine can be further developed for the APOLLO vehicle.

6. Servicing Equipment

The necessary servicing equipment for supplying liquid oxygen, liquid nitrogen, liquid propellant, cooling air, gaseous nitrogen - high and low pressure, etc., to support the hangar tests will be provided. The type of equipment will be similar to that used in past re-entry and space vehicle programs by GE-MSVD. In addition, the equipment that will be utilized in the Mercury Program will be investigated to determine the extent that this equipment can be used for the APOLLO vehicle.

7. System Checkout

With the completion of the tests on the subsystem, leak tests, weight and balance checks, etc., an electrical check will be made of the entire vehicle system — again utilizing the vehicle checkout console. These tests will check continuity of the electrical system as well as inter-action and compatibility of the subsystems with each other. It may be that some of the tests and checks made on a subsystem basis will be repeated at this time to determine the effects of interaction. The APOLLO space vehicle is now ready to be transported to the launch pad for Flight Readiness Firing Tests.

LAUNCH PAD-FLIGHT READINESS FIRING TESTS

From the hangar, the APOLLO space vehicle will be transported on the transport vehicle to the launch pad for the flight readiness firing tests. These tests are to prove mechanical and electrical compatibility of the APOLLO space vehicle with the

SATURN Booster and the launch pad cabling and electrical and mechanical ground launch support equipment. This will also point out any problems in the operation of vertically mating the APOLLO space vehicle to the SATURN Booster. All of the launch pad ground support equipment will be utilized prior and during these tests.

Before the static firing tests are started, an APOLLO space vehicle simulator will be operated to check the launch pad cabling and electrical ground equipment. Mechanical compatibility will be confirmed upon mating the space vehicle to the booster. Electrical compatibility will be confirmed using the space vehicle operating console and the electronic subsystem test set. The space vehicle will be serviced by the air-conditioning unit and other servicing equipment, as necessary.

During the actual static firing, all one-shot devices, less explosives, will be used. This test will confirm over-all space vehicle and booster compatibility using internal power. All of the recorded data received during the static firing will be sent to the hangar area for reduction and analysis. At the completion of the static firing tests, the space vehicle will be disassembled from the booster, lowered from the gantry, mounted on the transport vehicle, and returned to the hangar.

FIELD HANGAR-FINAL HANGAR CHECKOUT

Upon returning to the hangar following the flight readiness firing tests, an inspection will be conducted on all subsystems. Equipment will be provided in the hangar area for all requirements of maintenance, assembly and disassembly of the space vehicle down to the black-box component level. Following the inspection, leakage tests on the space vehicle plumbing and cabin areas will be repeated. Any leaks, found at this time, will be carefully investigated to determine the cause of the leak, and to insure that it will not recur under actual launch conditions. After these final preparations, a final hangar confidence test will be conducted, using the space vehicle checkout console and the electronic subsystem test set. The space vehicle will now be transported to the Explosives Installation Area.

EXPLOSIVES INSTALLATION AREA

The APOLLO space vehicle will be transported from the Hangar to the Explosives Installation Area for installation of any explosive devices and live rockets, and for performing any other operations which are considered to be a hazard to personnel safety by the Range Safety Office. Confidence checks will be made which include applying electrical power to the space vehicle, insuring compatibility with the explosives installed. During this procedure, the transport vehicle will be suitably braced or tied down to prevent movement in case of accidental rocket ignition. After satisfactory completion of tests, the APOLLO space vehicle will be transported to the launch pad with all safety circuits suitably monitored.

LAUNCH PAD- LAUNCH OPERATION

At the launch pad, the space vehicle will be vertically mated to the Booster with the same procedure and in the same sequence as was followed for the flight readiness firing tests. All ground servicing units will be connected to the space vehicle. Countdown is initiated. External power will be applied through the ground power system and, with the operating console and electronic subsystem test set, a test crew will conduct the operations in the cabin area and check all visual displays. This will include:

1. Check-off of crew's control and instrument panels.
2. Confidence checks of all subsystems.
3. Checkout of the Instrumentation and Communication Subsystem via open loop transmission.
4. Monitoring of all explosive and rocket firing circuits.

The space vehicle is now ready for the flight crew. With the flight crew in the vehicle, the countdown will now be continued to launch. The prelaunch checks will now include:

1. A repeat of the checks made by the test crew.
2. The space vehicle will be sealed and the cabin area purged.
3. A qualitative leak check of the sealed cabin area using the helium probe.
4. Top-off all space vehicle tanks - liquid oxygen, liquid nitrogen, liquid propellant, etc.
5. Continue pre-flight checks with crew.
6. Clear and move gantry.
7. Make final electronic subsystem check.
8. Flight crew moves into final position for launch.
9. Switch to internal power and service.
10. Disconnect all ground service connectors.
11. All systems ready.
12. LAUNCH!

Following missile "lift off" from the launch pad, the monitoring and command functions will be transferred to the Central Command Center at AMR. To facilitate quick observation of the space vehicle performance, a post launch monitoring set will be provided to be used in conjunction with the telemetry receiving equipment at the Command Center.

SEARCH AND RECOVERY AREAS

As a first cut, and until the vehicle trajectories and landing sites become more definitized, it is assumed that the search and recovery areas for the APOLLO program will be included in those called for in the Mercury Program, as well as land recovery areas such as Edwards Air Force Base. Equipment that will be in existence

from the Mercury Program, for both the search and recovery phases, will be utilized as much as possible. Equipment in existence from other Programs will also be used to the fullest. New search and recovery equipment will only be developed specifically for the APOLLO program if no other existing equipment is, or will be, available for use. See next section, "2. Search and Recovery", for further details regarding this phase.

GROUND TRACKING AND COMMUNICATION AREA

See Section "Search and Recovery Areas", page XIV-11. Same principles will apply in this area.

3. SEARCH AND RECOVERY

Prime intent will be to utilize existing site facilities to the fullest extent possible. For a ballistic-type re-entry vehicle, assuming correct geographical location prevails, minimum or no facility change is anticipated. If additional sites are required, reproductions of the originals will probably be constructed. However, should the re-entry vehicle be of the glide variety, additions in the form of runaways and arresting gear will necessarily be added to all planned sites.

The termination phase of an APOLLO mission will be either normal, abort, or contingent in nature. Touchdown potential, therefore, exists for a diverse combination of locations/conditions and, as such, all areas will necessarily have to be considered when evolving and implementing the search and recovery procedure. Possibilities include:

A. Primary prepared ground landing site

B. Secondary prepared ground landing sites

C. Unprepared ground landing sites

   1. Mountainous terrains (crevices, cliff-side, high altitude plateau, etc.).
   2. Swamps or marshes.
   3. Snow or glaciers.
   4. Jungles.

D. Calm sea (waves 6 feet, maximum)

E. Heavy sea (anything over 6-foot waves)

Coupled with these possibilities is any crew and/or re-entry vehicle disability which must be factored into the situation.

Initially it is felt that the following procedures and equipment possess the highest degree of feasibility and probability of success and efforts will proceed accordingly.

PRIMARY PREPARED GROUND LANDING SITE (PROBABLY THE MOST DESIRABLE RECOVERY)

1. Ballistic re-entry vehicle — At touchdown the re-entry vehicle could potentially become enshrouded in the parachute(s). Disentanglement could be accomplished either by the astronauts or base crewmen. Crew and data removal would be effected immediately via helicopter while re-entry vehicle retrieval and transport could be done in relative leisure.

Eventually, a hoist-equipped helicopter would attach to an re-entry vehicle pickup fitting and transport the entire assembly to the operations area. At this point a handling ring would be attached to the original re-entry vehicle mounting hard points. A portable crane/load balancer could then be utilized to lift the assembly and mate the trunnions of the handling ring with appropriate receptacles in a portable handling stand (Type I mobility).

2. Glide re-entry vehicle — During landing roll, the re-entry vehicle (possibly the alighting wheels or skids) would engage a cable-anchor chain arrangement; the gradual addition of drag weight supplying smooth deceleration. Crew, data, and re-entry vehicle will be retrieved and transported to the operations area in the same sequence and manner as for the ballistic type. Possible additional equipment required might include portable dollies to be placed under the re-entry vehicle alighting skids to provide some mobility.

SECONDARY PREPARED GROUND LANDING SITES

The requirements and procedures for these facilities will be the same as those for the primary installations.

UNPREPARED GROUND LANDING SITES

Generally, a landing resulting from some contingency inherently possesses a higher probability for personnel injury and/or vehicle damage. (This characteristic is somewhat reduced in the case of a glide vehicle since more maneuvering to avoid local obstacles is possible.) Therefore, not only is the vehicle located in a potentially awkward, inaccessible area, but little or no active help may be forthcoming from its occupants.

The tracking-communication-computer link will provide forwarning of deviation from the planned impact area. Long range search aircraft, in ready status, would be dispatched to the predicted impact area. A search pattern would be flown and, upon locating the re-entry vehicle, the situation would be appraised via communication and/or visually.

Supplies and supplementary equipment would be dropped (assuming life has been detected); paramedics may or may not be utilized depending upon circumstances.

Crew, data, and vehicle will be removed in that order with prime emphasis devoted to the crew and then the data. The demonstrated versatility of helicopters makes them ideally suited to usage for this application.

CALM SEA

1. Ballistic re-entry vehicle — After impacting, stabilization, and flotation, the re-entry vehicle could potentially become enshrouded in the parachute(s). Disentanglement could be accomplished either by the astronauts or subsequent rescuing personnel.

Search aircraft, upon locating the vehicle, would maintain surveillance and direct rescue activity to the area. Search helicopters would immediately commence rescue operation. Astronauts and data would first be removed by means of a hoist-sling/seat

arrangement. Vehicle recovery would then be initiated. The procedure involved could be similar to that successfully employed by General Electric in the recovery of the RVX-2A nose cones.

A back-up flotation buoy would be lowered into the water and attached to the re-entry vehicle by a helicopter crewman, who would also deactivate any pyrotechnic devices. A surface ship would "heave to" in the area and lower a powered whale boat over the side. Crewmen working from the small boat would attach steadying lines to the re-entry vehicle and connect the ship's boom-load balancer combination to the capsule pick-up fitting. The assembly would then be hoisted out of the water and directed over the deck where the steadying lines would be manned by deck hands. The vehicle would then be lowered onto a sandbag cushion where, with partial tension maintained in the boom cable, the buoy is removed and a handling ring is attached to the original re-entry vehicle mounting hard points. The boom would then lift the assembly and mate the trunnions of the handling ring with appropriate receptacles in a handling/shipping stand which could be lashed to the deck.

2. Glide Re-entry Vehicle — The entire procedure outlined for the ballistic type would be applicable for this case. However, the sandbags and handling/shipping stand would not be required since the re-entry vehicle could be lashed directly to the deck.

HEAVY SEA

The same general search and recovery techniques would be carried out as for a calm sea state. However, participation of certain units and consequent results will be directly proportional to the degree of prevailing natural violence — high wind, clouds, rain, waves, etc.

HELICOPTERS

These vehicles are proven workhorses of a rescue mission. Although there are problems involved in their use, sufficient operational experience has been gained to insure a high reliability in recovery operations. They would provide the fastest recovery of

TABLE XIV-3-1. HELICOPTER DATA

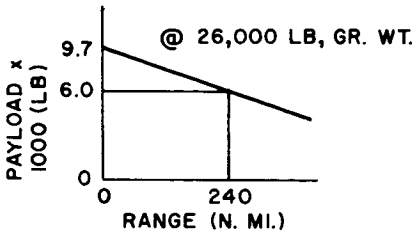
1. H-25/HUP-2 (Vertol)
   Cruise - 80 MPH
   Range - 340 miles

   Useful load - 400 lb.
   Accommodations - 4 passengers or 3 stretchers.

2. H-21 (Vertol)
   Cruise - 98 MPH
   Range - 115 Miles

   Useful load - 3000 lb.
   Accommodations - 20 passengers.

3. 107/YHC-1B (Vertol)
   V max - 160 MPH



4. H-43/HOK-1 (Kaman)
   Cruise - 75 MPH
   Range - 220 miles

   Useful load - 600 lb.
   Accommodations - 3 passengers or 2 stretchers.

5. S-56/H-37/HR2S (Sikorsky)
   Cruise - 125 MPH
   Range - 200 Miles

   Useful load - 7000 lb.
   Accommodations - 36 troops or 24 stretchers.

6. HSS-1/HUS-1 (Sikorsky)
   Cruise - 98 MPH
   Range - 280 Miles

   Useful load - 3000 lb.
   Accommodations - 12-18 troops.

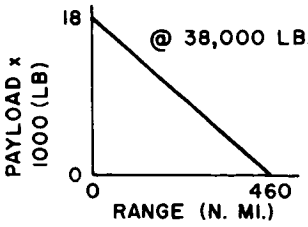
7. S-60 (Sikorsky)
   Can lift at least 2 tons

8. S-61/HSS-2/HRS-2 (Sikorsky)
   Useful load of at least 6000 lb.

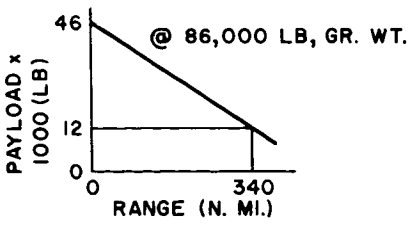
9. S-62 (Sikorsky)
   Cruise - 92 MPH

   Useful load - 1840 lb.
   Accommodations - 8-12 passengers

10. S-64 (Sikorsky)
    Cruise - 109 MPH



11. Proposal (Sikorsky)
    Cruise - 98 MPH



crew and data and under some of the conditions delineated above would afford the only practical, safe method of recovering crew, data, or capsule. Several classes of these machines appear likely for consideration and currently available data for them is shown in Table XIV-3-1.

SPECIALIZED RETRIEVAL CONCEPTS

With the increasing tempo of space explorations which require recovery vehicle attempts and especially with the impending inclusion of humans in the vehicles, it might be logical to consider that specialized retrieval concepts designed specifically for the task would be contrived at this time. Retrieval methods currently utilized are implemented with equipments carried on military inventory — equipments not specifically designed for the task involved. As such, some phases of recovery are becoming marginal and astronaut survival may be unduly jeopardized.

4. FACILITIES

As a first cut, the facilities required for the APOLLO program at the launch areas, search and recovery areas, and the ground tracking and communication areas will be similar to those provided for the Mercury program. However, if the basic vehicle configuration becomes a glide-type re-entry vehicle, then additional and expanded facilities will become necessary. For all primary and secondary landing areas, a runway or landing strip will be required. The same type of arresting gear now included on many Air Force and Navy installations for slowing down high-speed landing aircraft will also be required at these landing areas for glide-type re-entry vehicles.

CHAPTER XV

ELECTRICAL POWER

XV ELECTRICAL POWER

1. PRELIMINARY SYSTEM COMPARISON

A number of possible systems are in competition to provide electrical power for the APOLLO Mission. Depending on their source of energy, they fall in three categories: solar, chemical, and nuclear. Based on information presently available, no one of these systems can be absolutely discarded at the present time. In order to ascertain the effects of any operating or conceived power system on the proposed APOLLO Mission requirements, a set of specifications (Missile and Space Vehicle Department Specification No. 2855) has been made available to numerous power system vendors. These have been forwarded to the companies listed in Appendix I. Within a month's time we expect to receive study results from participating companies for evaluation. We feel that in this manner the best power system can be selected to meet the mission requirements.

Seven power system types have been taken into consideration:

1. Photovoltaic/Chemical Storage
2. Solar Collector/Dynamic System
3. Solar/Thermionic System
4. Hydrogen-Oxygen Combustion Systems
5. Open-Cycle, Hydrogen-Oxygen Fuel Cells
6. Nuclear Reactor—SNAP II Systems
7. Nuclear—Thermal Multiplier Systems

Some of the advantages and disadvantages of each of these systems are listed in Table XV-1-1. State-of-the-art development is estimated in Table XV-1-2.

TABLE XV-1-1. GENERAL SYSTEM COMPARISON

	System	Advantages	Disadvantages
1.	Photovoltaic/ Chemical Storage	Meets mission requirements, high reliability due to system simplicity and minimum of moving parts; requires no additional radiator area; proven application in satellite programs.	Necessity of orienting solar array during flight duration; provision of secondary power supply during dark time period of the mission.
2.	Solar Collector/ Dynamic System	Meet mission requirements; thermal storage technique can eliminate need for secondary power source for dark time periods.	Much stricter orientation tolerances for collector; complexity of heat transfer system and energy conversion unit provide greater opportunity for malfunction and decrease reliability of system as a whole; unproven application to space flight conditions.
3.	Solar Collector/ Thermionic	Static System	Strict orientation requirements for collectors; thermionic converter development time.
4.	Hydrogen-Oxygen Combustion Systems	Possibility of integration with life support and attitude control systems.	High weight of fuel supply in addition to fuel storage requirements; starting capability in a space environment; no demonstrated performance on low-power, long duration dynamic systems of this type; low density of hydrogen leads to high volume requirements.
5.	Hydrogen-Oxygen	Static system; possible integration into other vehicle systems.	Projected S.F.C. in order of 1.2 lbs/kw-hr for future application places system outside APOLLO application unless suitable integration with other systems is made.
6.	SNAP II Reactor/ Turbo-Generator	Inherent growth potential due to reactor as a heat source. Experimental reactor had demonstrated 1,000 hours life at	Shielding weight exceeding 2,000 pounds required for APOLLO mission. Power source not yet integrated to conversion unit. Compexity of reactor-boiler-conversion-unit-radiator system

TABLE XV-1-1. GENERAL SYSTEM COMPARISON (Continued)

	System	Advantages	Disadvantages
6.	(Continued)	operating temperature and pressures.	leads to doubtful reliability in a short period. Additional power supply needed for launch and re-entry.
7.	Thermal multiplier/ Thermionic Converter	Static system, inherent reliability.	Same shielding requirement as for SNAP II Reactor. Thermionic converter development time.

TABLE XV-1-2. STATE-OF-THE-ART OF VARIOUS SYSTEMS

1. Photovoltaic/chemical energy storage – safe and reliable application in satellite programs - concentrating mirrors for increased power output in development.
2. Solar collector/dynamic system – a prototype of this system is expected to be operational by 1963-1964.
3. Solar/thermionic – development phase – efficient converters with applicable life not expected until 1963.
4. H_2O_2 combustion/dynamic – systems for low-power production still in study phase.
5. Open-cycle fuel cells – heat and water removal problems extend development of these systems to 1962-1963.
6. SNAP II – Reactor/turbo-generator system – proposed flight target is mid-1964.
7. Thermal multiplier – General Electric proprietary development – work has not progressed beyond the conceptual stage.

Key problem areas exist for some of the systems for application to the APOLLO Mission. A power system weight not exceeding 1,000 pounds maximum has been set by the vehicle characteristics, to supply a nominal electrical load of 3.0 kilowatts.

The photovoltaic/chemical storage system can be designed to meet both power and weight requirements for APOLLO. Orientation requirements for the solar array are much less rigid than other solar devices being in the order of plus or minus 5 degrees.

In a solar collector/dynamic system, it is possible that the weight requirement can be met. The additional complexity of rotating machinery and the more stringent orientation requirements in comparison to the photovoltaic array tend to reduce system reliability and increase weight.

The attractiveness of a static system is demonstrated by a solar/thermionic system. However, the use of thermionic converters for APOLLO is limited by the state-of-the-art since considerable development work to extend operating life is necessary. This system also has the same strict orientation requirement as the dynamic system previously mentioned.

Open-cycle hydrogen-oxygen combustion systems are penalized by weight and volume requirements of the fuel. Fuel weights alone exceed 2,000 pounds, but it is possible this system could be redeemed by integrating the cooling capability of cryogenic fluids into other vehicle systems such as environment control and crew support.

Open-cycle hydrogen-oxygen fuel cells are in the development stage and may have application to this mission. Projected specific fuel consumptions in the order of 1.2 pounds/kw-hr point to high fuel weights with doubtful reliability. The possibility of integration with other vehicle systems is present, although the cooling capability is reduced because of the smaller hydrogen requirement.

A nuclear system of the type laid out in Figure XV-1-1 for the APOLLO Mission is attractive from a growth approach. The SNAP II Reactor has undergone performance test and demonstrated capability to supply the required power. Although the reactor and power conversion unit weight is compatible with APOLLO requirements, manned flight demands radiation protection. Shield weight exceeding 2,000 pounds would have to be added to basic system weight for the APOLLO mission.

The thermal multiplier consists of using a sub-critical reactor to multiply heat emitted from a small radioactive isotope source. The neutron source initiates and

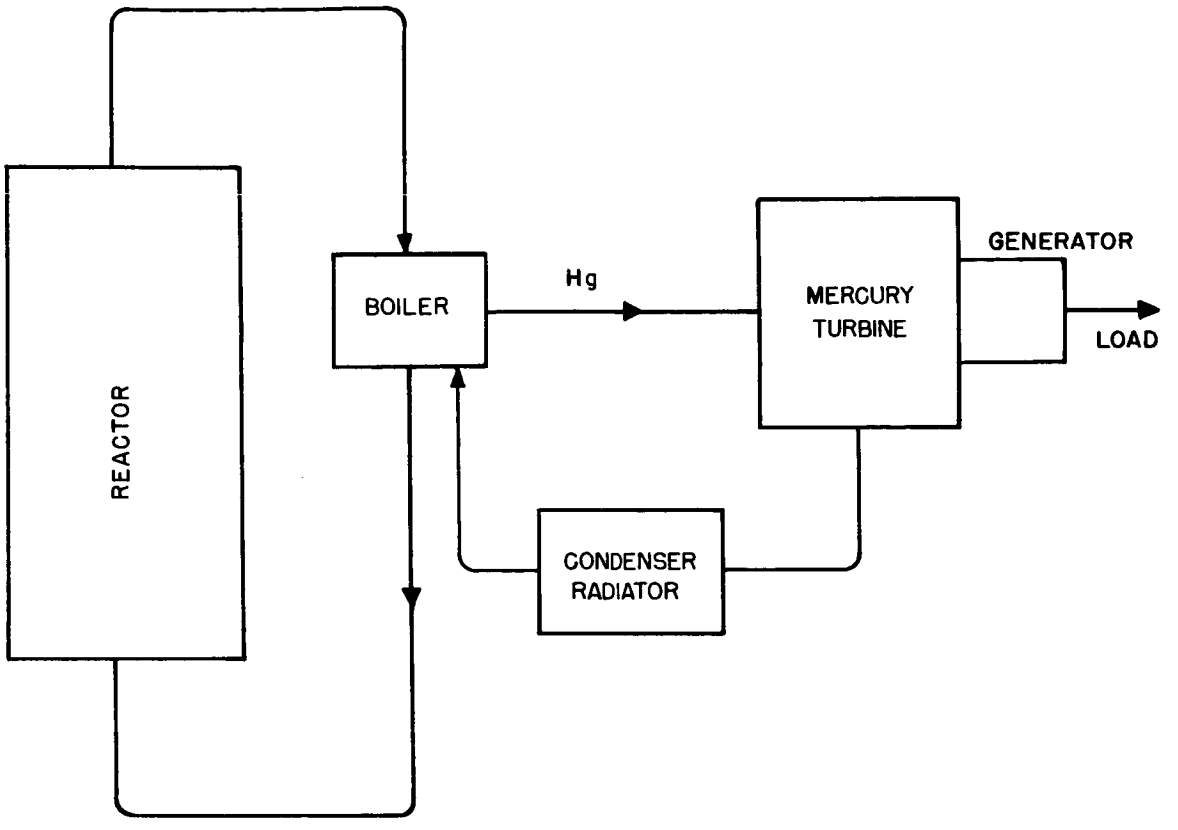


Figure XV-1-1. Nuclear system —SNAP II version

stabilizes a fission reaction. Heat is transmitted to thermionic converters mounted on the reactor surface to supply electrical energy. This nuclear system shown in Figure XV-1-2 has the advantage of being a static device uniquely adaptable to low-power production. Development has not progressed beyond the conceptual stage and the shield requirement would be similar to that for the SNAP II Reactor.

Since a power supply system for manned-space flight should be inherently reliable and as simple as possible, it is felt that the course of action we have taken in inviting outside participation will result in the choice of the best system available at the time. During the course of our investigation, it is our intention to design a photovoltaic/chemical storage power system to use as a basis for comparison with other proposed systems. A preliminary design of such a system is appended.

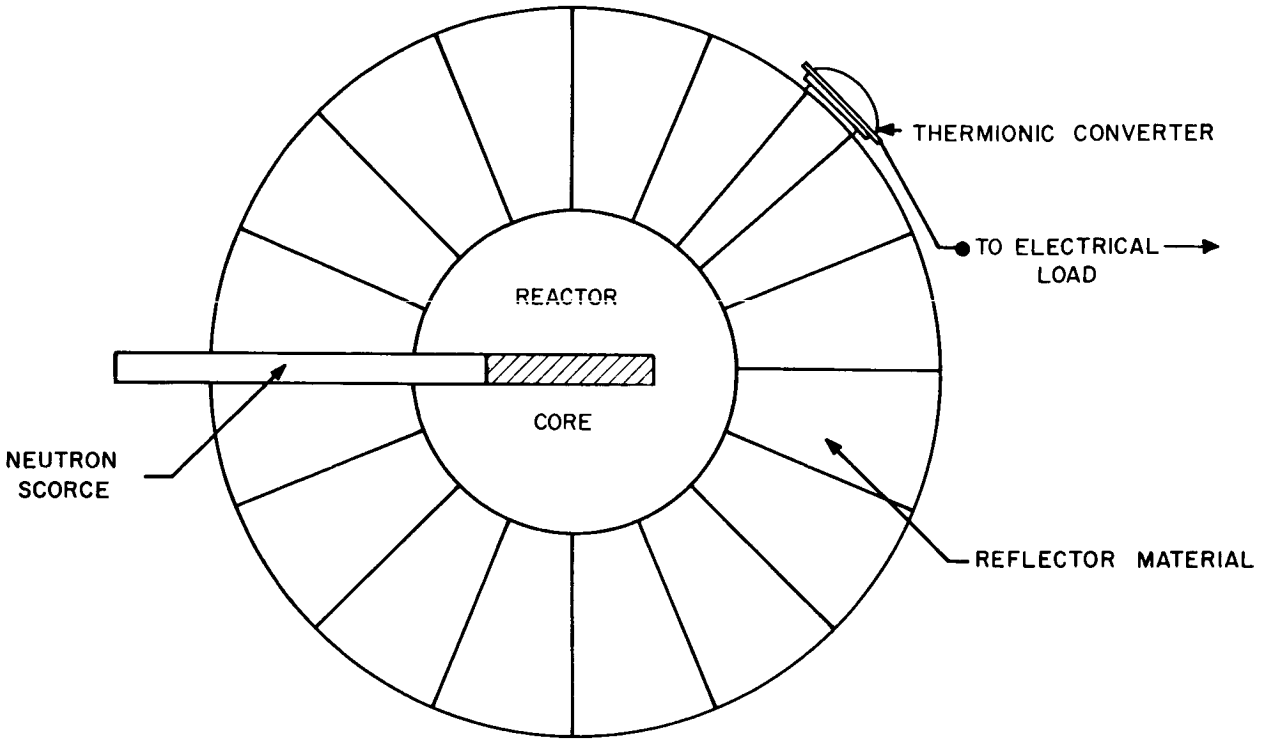


Figure XV-1-2. Thermal multiplier

2. SOLAR PHOTOVOLTAIC – FUEL CELL SYSTEM DESIGN

This solar photovoltaic design of the Electrical Power and Distribution subsystem is based upon the lunar orbiting mission requirement. The requirements of high reliability, low weight and varied performance seem to establish this as the limiting case of any mission.

The mission is broken into six phases. Each phase has an electrical load determined by the vehicle equipment required to perform during that phase. A summary of the six phases is:

1.	Prelaunch, launch to staging end	2300 watts	15 minutes
2.	Earth-to-Moon trajectory	2360 watts	3.5 days
3.	Lunar orbiting	2500 watts	7 days
4.	Moon-to-Earth trajectory	2360 watts	3.5 days
5.	Re-entry to Earth	1905 watts	15 minutes
6.	Earth surface recovery	100 watts	72 hours

The lunar orbiting period requirements establish the design criteria for the solar array and energy storage. At an average lunar altitude of 500 nautical miles the orbit period would be 220 minutes with 45 minutes of it in the Moon's shadow. The dark period determines the battery or fuel cell size. The cycle efficiency of the storage media and the light-to-dark ratio establishes the increase in solar array size over the vehicle system load. Due to the large amount of power required (4 kilowatts), conventional solar photovoltaic configurations would be unwieldly. This preliminary design is therefore based upon concentrating-type solar photovoltaic systems using a regenerative hydrogen-oxygen fuel cell for energy storage.

Figure XV-2-1 shows a block diagram schematic of the power subsystem. The solar array and fuel cell are divided into eight identical modules. Each module operates independently and cannot be degraded by a failure in any other module. Four modules are paralleled per bus. The two buses are independent. Critical vehicle loads, i.e.,

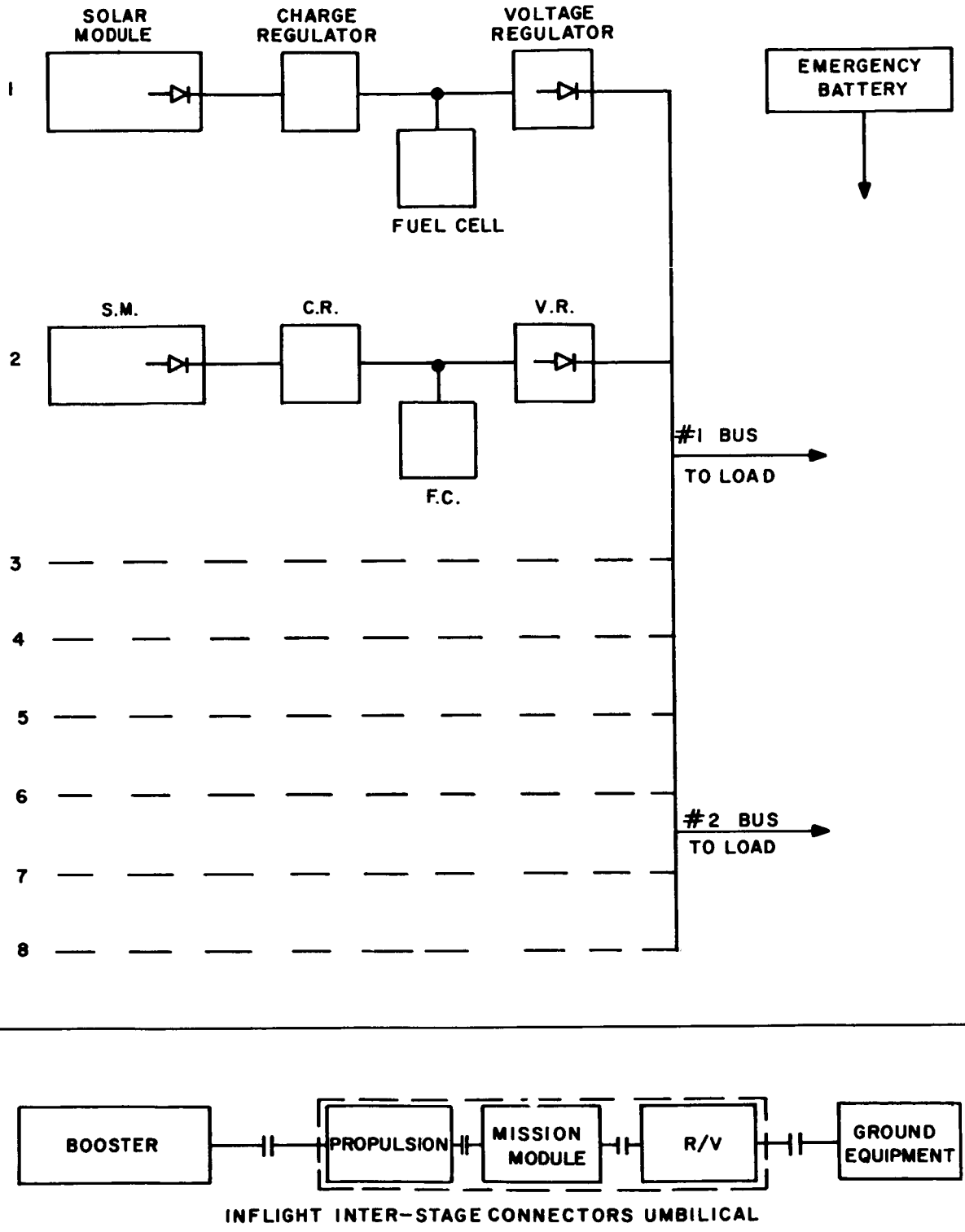


Figure XV-2-1. Electrical power and distribution system—block diagram

those which would cause catastrophic mission failure, can be switched between buses in the event of load or power supply failure on the original bus. This provides extremely high reliability, since only a maximum of 12.5 percent of the full-load power could be lost per unit power system failure.

The solar photovoltaic collector consists of a Cassegrainian-type parabolic concentrator, Figure XV-2-2. Geometric parameters are given for four configurations, i.e., one, two, three or four concentrators. Each system has the same total power capability. The four configurations allow maximum freedom for the vehicle design and control subsystem optimization. The concentrator would be deployed after booster separation and oriented to the Sun. During the Earth-to-Moon trajectory it would supply the vehicle system power as well as replenishing the fuel cell energy used during launch. The maximum requirements of 4000 watts during orbit would be supplied during the light periods and the fuel cell would supply the vehicle load during shadow periods. During the Moon-to-Earth trajectory the solar array would again supply 100 percent of the vehicle load.

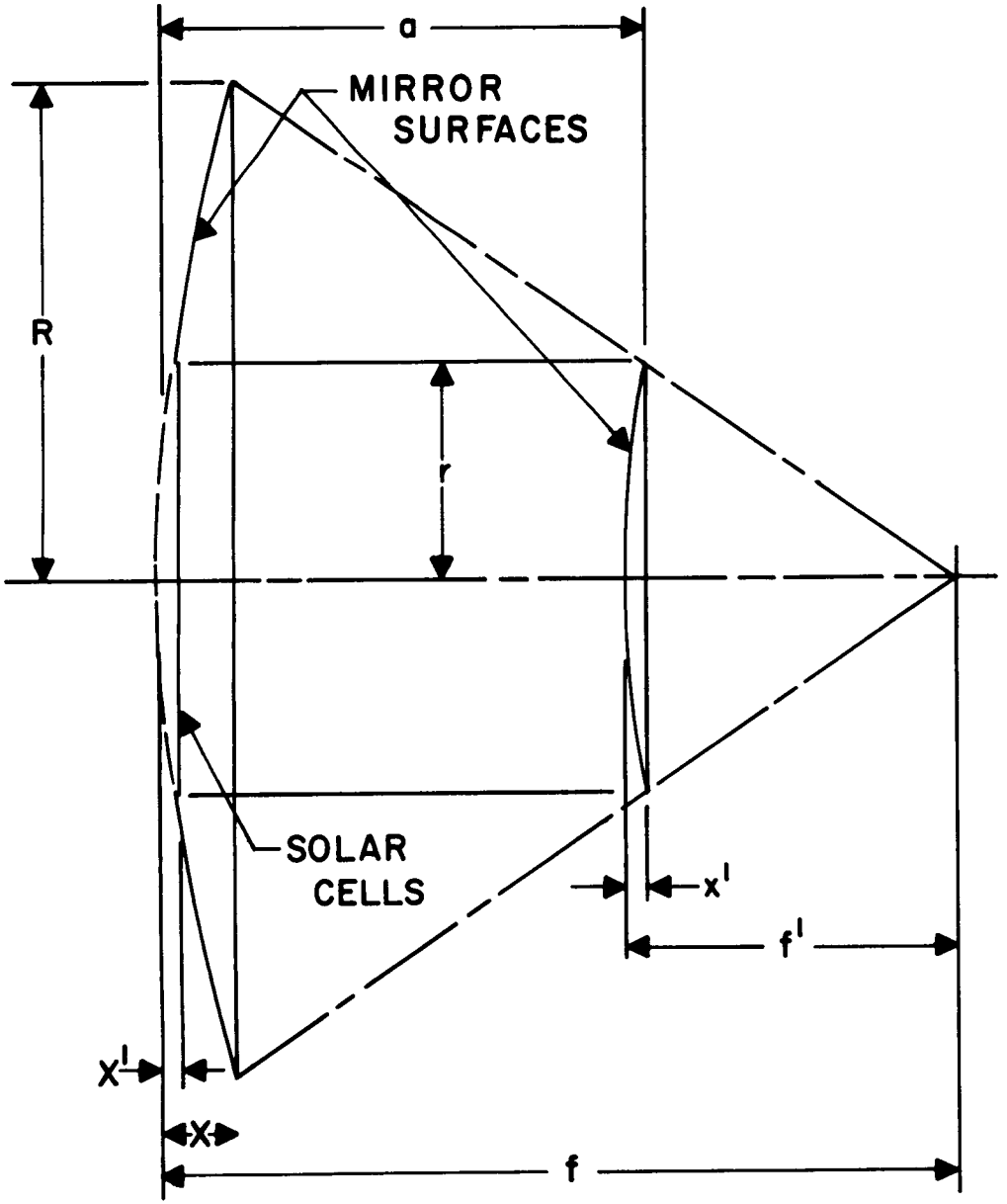
Just prior to Earth re-entry, the solar collectors would be jettisoned or expended as high-altitude drag devices. The fuel cells would provide power during re-entry and recovery.

An emergency battery (primary-type, zinc-silver-oxide) is available in addition to the solar array - fuel cell system. This battery will supply a limited amount of emergency power during the mission and/or during recovery. It would not be required for any normal mission.

Table XV-2-1 provides a summary of the power and distribution subsystem weight and volume. All inverting or converting equipment other than that listed will be provided within each of the using subsystems.

At this time, the solar photovoltaic - fuel cell system appears to be the most reliable, lightest in weight, most versatile and most advanced in its state-of-the-art of all space power systems.

CONFIDENTIAL



NO. OF COLLECTORS				
	1	2	3	4
R	18.85	13.3	10.9	9.44
X	3.21	2.26	1.94	1.68
X'	0.46	0.32	0.277	0.24
f	27.65	19.58	15.31	13.26
r	7.13	5.04	4.12	3.57
f'	10.45	7.39	5.83	5.05
x'	1.22	0.86	0.72	0.63
a	18.45	13.0	10.2	8.83

DIMENSIONS IN FEET

Figure XV-2-2. Solar cell concentrator system

CONFIDENTIAL

TABLE XV-2-1. ELECTRICAL POWER AND DISTRIBUTION

WEIGHT AND VOLUME SUMMARY

Components	Volume Total	System Design Weights I	II	III	IV
1. Fuel Cell - 8 Units	3.0 ft^3	190	190	190	190
2. Emergency Battery	0.9 ft^3	103	103	103	103
3. Charge Regulator - 8 Units	0.8 ft^3	32	32	32	32
4. Voltage Regulator - 8 Units	0.8 ft^3	32	32	32	32
5. Inflight Connector	—	5	5	5	5
6. Interstage Connectors (2)	—	10	10	10	10
7. Umbilical - (Flight Half)	—	5	5	5	5
8. Vehicle Harness	—	45	45	45	45
9. Solar Array	—	425	450	460	460
I - One Collector					
II - Two Collectors					
III - Three Collectors					
IV - Four Collectors					
Total Systems	5.5 ft^3	847	872	882	882

3. APPENDIX I

Preliminary specifications have been forwarded to a number of vendors in the space power field. A list of these vendors follows. Asterisks indicate those who intend to submit study results for our evaluation.

Allison Division, General Motors Corporation*
Vickers, Inc.*
Marquardt Corporation*
Aerojet General Corporation*
Atomics International
Sundstrand Aviation
Bendix Corporation
Electro Products Laboratories
Aircraft Accessory Turbine Dept., General Electric Co.*
Flight Propulsion Laboratory Dept., General Electric Co.*
Hamilton Standard Divisions U.A.C.
Pratt & Whitney Aircraft
Lockheed Corporation
Westinghouse Electric Corporation
Stratos Div., Fairchild Aircraft
Thompson Ramo-Wooldridge, Inc.
Boeing Aircraft
Ionics, Inc.
Radio Corporation of America
Hoffman Electronics
Philco Corporation
Goodyear Corporation
North American Aircraft
Northrop Corporation
Chance Vought Aircraft
Space Technology Laboratory

CHAPTER XVI

VEHICLE DESIGN

XVI. VEHICLE DESIGN

1. INTRODUCTION

This section contains the design description for selected semi-ballistic and glide re-entry vehicle configurations. Furthermore, configuration and arrangement studies on complete vehicles as well as re-entry vehicles are summarized for both concepts, and selected over-all vehicles are described in detail. As a result of the latest trade-offs, the R-3 and Modular "D" configurations have been selected for further study and all other shapes will be dropped. Parallel analysis of these two re-entry vehicles give a comprehensive comparison of glide versus ballistic re-entry from the development and manufacturing aspects of the program, as well as from mission performance capability.

In general, the results of these studies show lighter weights for the D vehicles, and superior terminal maneuverability for the R-3 glide vehicle.

2. SEMI-BALLISTIC-TYPE VEHICLES

SUMMARY

General Design studies for the APOLLO Space Vehicle (S/V) have been conducted to develop an over-all configuration and general arrangement of Re-entry Vehicle (R/V), subsystems, and components, which will meet all mission requirements at the lowest practical system weight. Materials and types of construction considered are in current usage within industry, or currently in a development status compatible with the APOLLO delivery schedule.

For all semi-ballistic R/V shapes both integral and modular configurations were studied. In all cases, the modular arrangement, separate mission and command areas, resulted in a lighter over-all vehicle weight. This was due, primarily, to the integrated R/V requiring both structural and heat protection weight penalties chargeable to the larger wetted area, as opposed to the lower structural and heat protection weight of the small modular R/V plus the light weight of the separable mission module which does not re-enter.

Re-entry vehicle shapes classified by families as B, C, and D were studied, with the D shape having the lightest weight and best overall packaging efficiency as well as attractive aerodynamic performance characteristics. The variation between families is mainly one of "bluntness," and representative configurations of these families are given in Figures XVI-2-1 through XVI-2-3.

System weight studies based on present technology have resulted in preliminary weight estimates as shown in Section "Weight Analysis". The C configuration is virtually eliminated due to weight and the B and D shapes are approximately 10 percent heavier than the

specified 15,000 lb weight. It is predicted that by 1963, the system weight requirements will be met or bettered. In structural design alone limited usage of sheet beryllium, improved propellant case design, and design and fabrication refinements could easily account for 300 to 400 lbs. An additional weight serving of 100 to 300 lbs may be realized from improved heat protection materials developments. In all subsystems, miniaturization and improved designs could account for comparable weight reductions.

The selected semi-ballistic vehicle is the modular "D" arrangements described below.

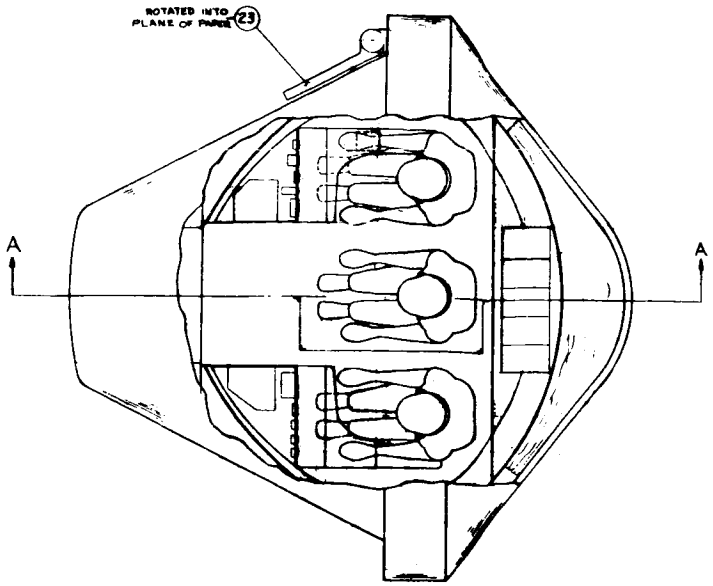
RE-ENTRY VEHICLE CONFIGURATIONS

The various semi-ballistic re-entry vehicles studied during the program are described below. In general, for the configuration study, the semi-ballistic vehicles utilize ablation heat protection systems and parachutes for landing.

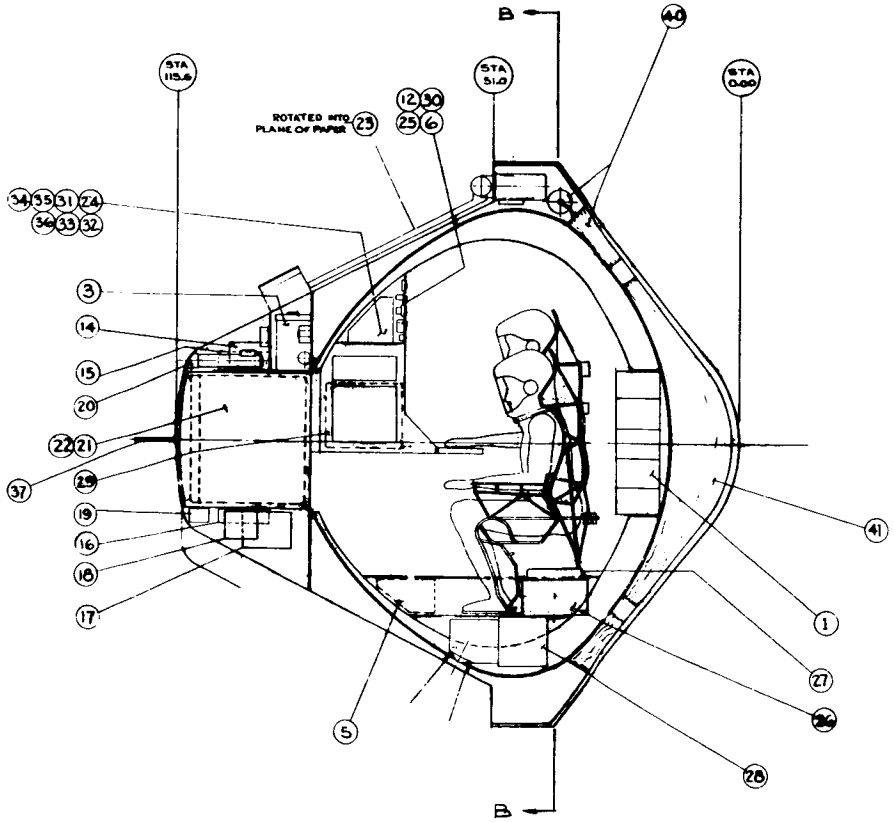
'B' Configuration

The "B" configuration, Figure XVI-2-1, is derived from the Mark 2 nose cone shape. As a command module, this particular configuration of R/V, designated B-2, weighs 5400 pounds, and has a W/C_DA of 59.5 lbs per sq ft. Hypersonic and subsonic L/D's are 0.29 and 0.58 respectively. Control is achieved by use of gas reaction jets for roll and a single flap for pitch and yaw. The vehicle is rolled to the proper orientation and the flap actuated for lateral and/or pitch control.

Heat protection during re-entry is provided by an ablation heat shield covering the entire outer surface. Internal structural temperatures are held below 300 F in order to utilize conventional aluminum or magnesium materials. The heat shield materials



PLAN VIEW

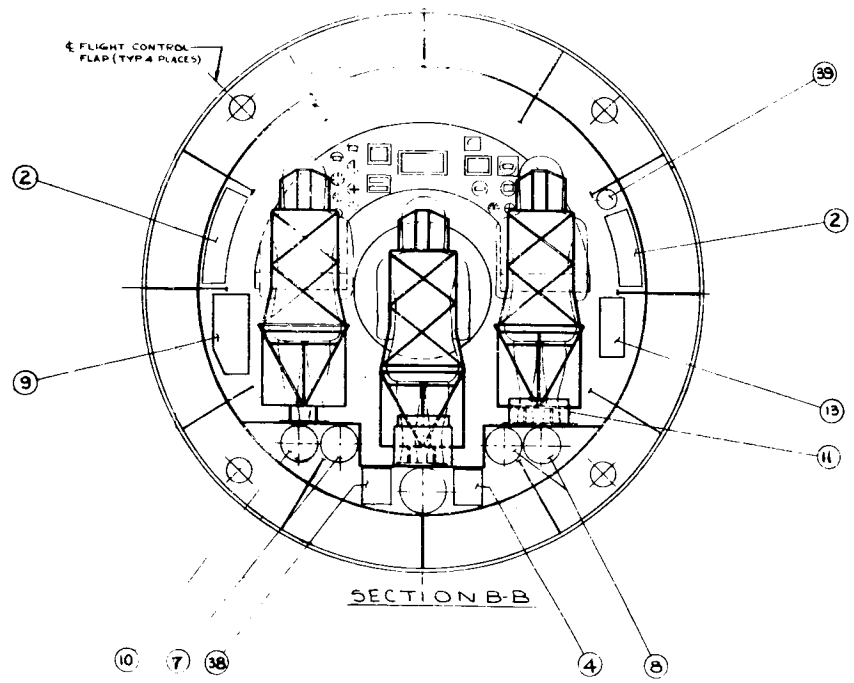


SECTION A-A

1 BATTERIES
2 FOOD & WATER (EMERGENCY
3 SURVIVAL EQUIPMENT
4 COMMAND CONTROL
5 PERSONAL EQUIPMENT
6 CONTROL CONSOLE DISPLAY PANEL (TEST & REPAIR CONSOLE)
7 O_2 SUPPLY (EMERGENCY)
8 DILUENT SUPPLY (EMERGENCY
9 CO_2 REMOVAL (PRIMARY INCLUDES NOXIOUS & TOXIC GAS REMOVAL
10 CO_2 REMOVAL (EMERGENCY
11 HEAT EXCHANGER (INC. WATER REMOVAL
12 FIRE CONTROL
13 SENSING & CONTROL
14 BEACON INCLUDES ELECTRONIC
15 CHAFF EJECTOR
16 FLARE
17 POWER SUPPLY (BATTERY)
18 PROGRAMMER
19 LIGHTS
20 SOFAR BOMB
21 MAIN CHUTE (INC. DYE MARKER & CHAFF)
22 BRAKE CHUTE
23 RE-ENTRY FLIGHT CONTROL FLAPS,
24 ACCELEROMETER
25 CLOCK
26 STABLE PLATFORM
27 PLATFORM ELECTRONICS
28 ASTROTRACKER
29 COMPUTER
30 CHRONOMETER
31 LUNAR TERMINAL GUIDANCE EQUIP.
32 RADAR ALTIMETER
33 NAVIGATION AIDS
34 400 MC TRANSMITTER & RECEIVER
35 DATA RECORDER
36 MULTIPLEXER
37 ANTENNA
38 VEHICLE CONTROL
39 [illegible] TUNNEL (RCV)
40 FLOTATION SYSTEM
41 IMPACT ATTENUATIO.. SYSTEM

DESIGN DATA

SURFACE AREA 2465 FT²
GROSS WT.



SECTION B-B

Figure XVI-2-1. Inboard profile, B-2 configuration

considered have been successfully flight proven on the Mark 3, RVX-2, and NERV programs, and represent a present design capability and know-how.

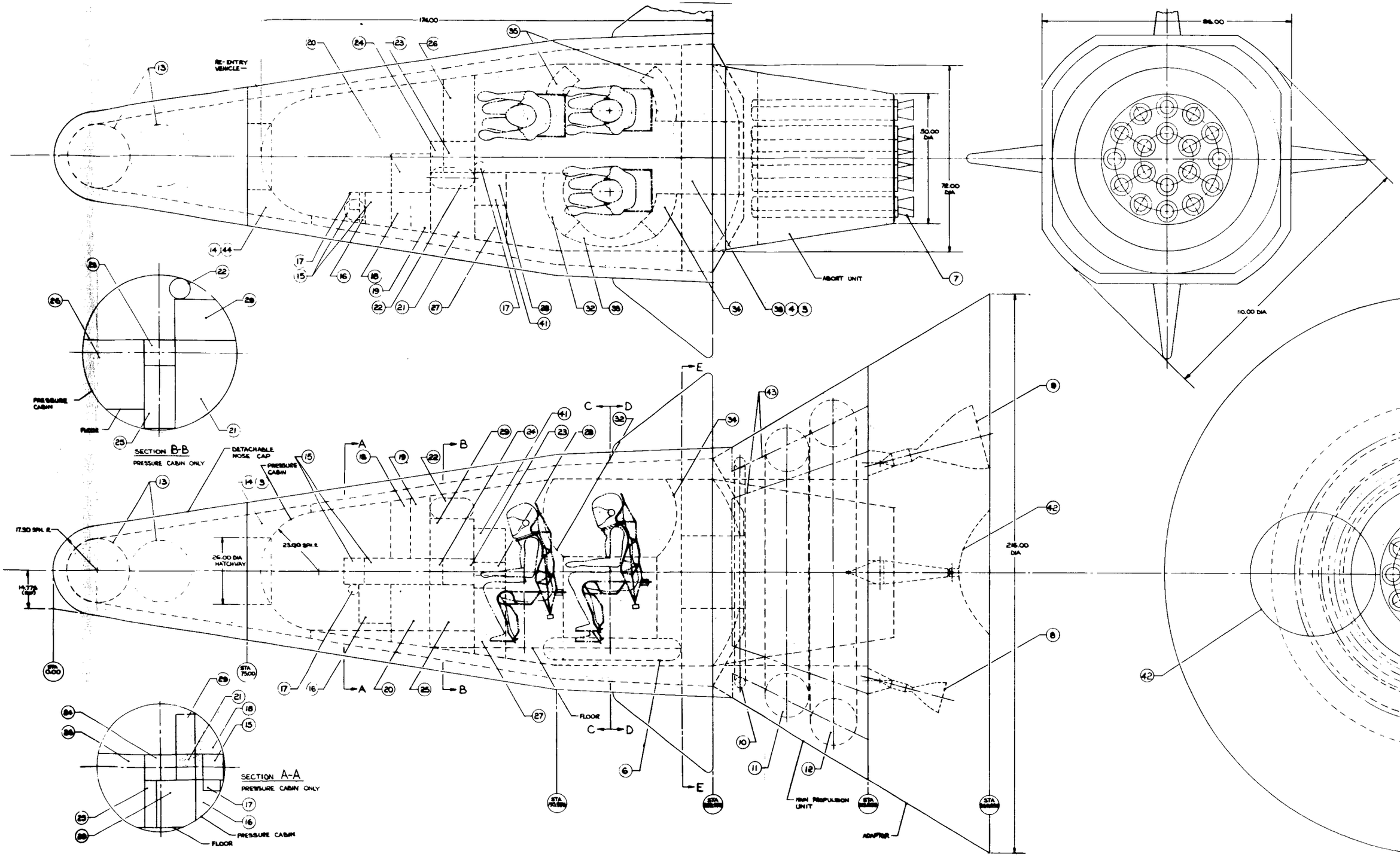
The three crew members are seated three abreast facing backward for the most desired orientation for acceleration, and a port may be provided for use after the re-entry phase if desired. Retardation and landing parachutes and recovery aids packaged in the R/V afterbody are deployed after re-entry. Landing impact is absorbed by an impact bag located behind the nose heat shield and extended after 'chute deployment.

'C' Configuration

The typical "C" configuration shown in Figure XVI-2-2 is similar to the RVX-2 sphere-cone shape. The pictured configuration, C-1, is one of the integral configurations studied. Ablation materials cover the entire nose and flare surfaces and control is by four movable fins on the aft flare section. Packaging of this shape is more difficult than the blunt "B" shape due to restricted usable volume in the nose, creating a problem in getting the CG sufficiently far forward for stability during re-entry. Crew members are seated two abreast in the aft location and 1 man forward. Weight of this configuration is 8000 pounds. Recovery 'chutes are located forward of a jettisonable aft cover, and landing impact may be absorbed by either crushable nose structure, or an impact bag. The "C" configuration is not considered desirable for the APOLLO application due primarily to consistently higher weights and difficulty of installation of the longer R/V which will be shown later.

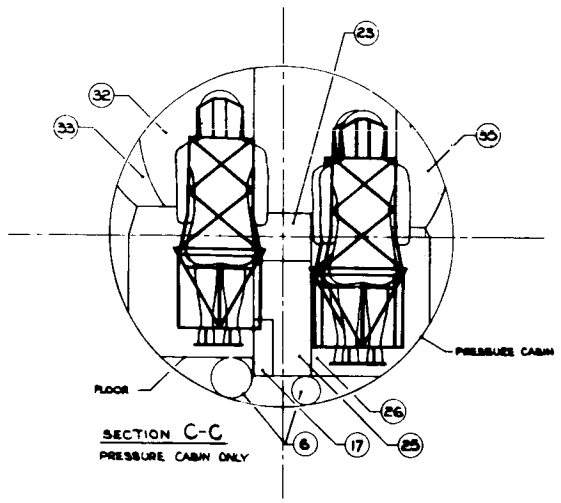
'D' Configuration

The "D" configuration, as represented by the D-2 of Figure XVI-2-3, represents the most desirable semi-ballistic shape studied. This shape has more usable internal volume than either the "B" or "C" shapes for a given base diameter and has basic characteristics and materials similar to the "B" vehicle. Weight for the D-2 is 4815

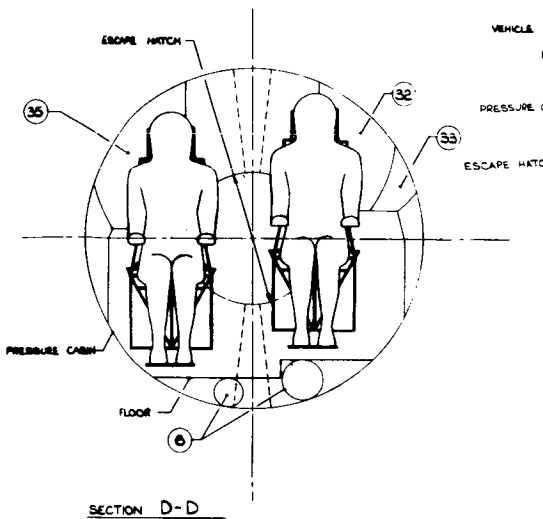

RE-ENTRY VEHICLE
ABORT UNIT
50.00 DIA
72.00 DIA
110.00 DIA
SECTION B-B
PRESSURE CABIN ONLY
DETACHABLE NOSE CAP
PRESSURE CABIN
FUSELAGE
26.00 DIA HATCHWAY
FLOOR
SECTION A-A
PRESSURE CABIN ONLY
MAIN PROPULSION UNIT
ADAPTER
215.00 DIA

FOLD-OUT #1

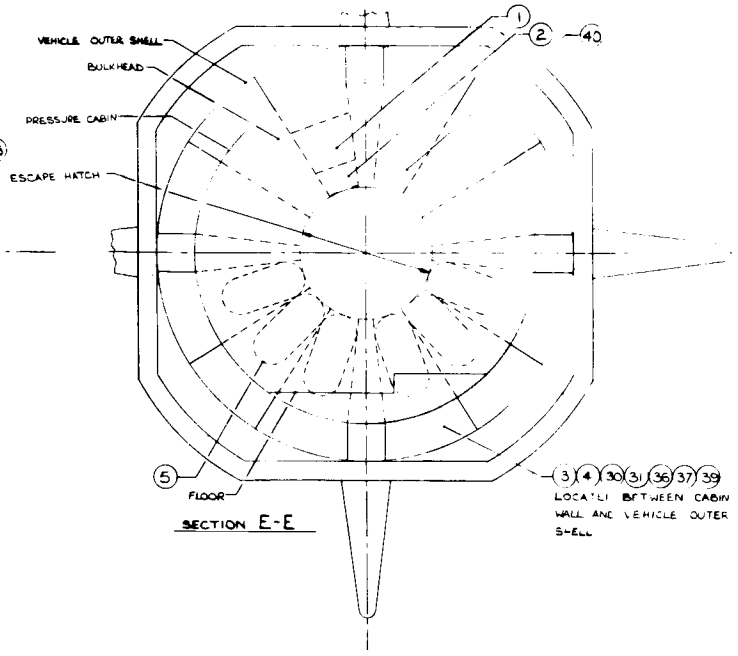
CONFIDENTIAL



SECTION C-C
PRESSURE CABIN ONLY



SECTION D-D
PRESSURE CABIN ONLY



SECTION E-E

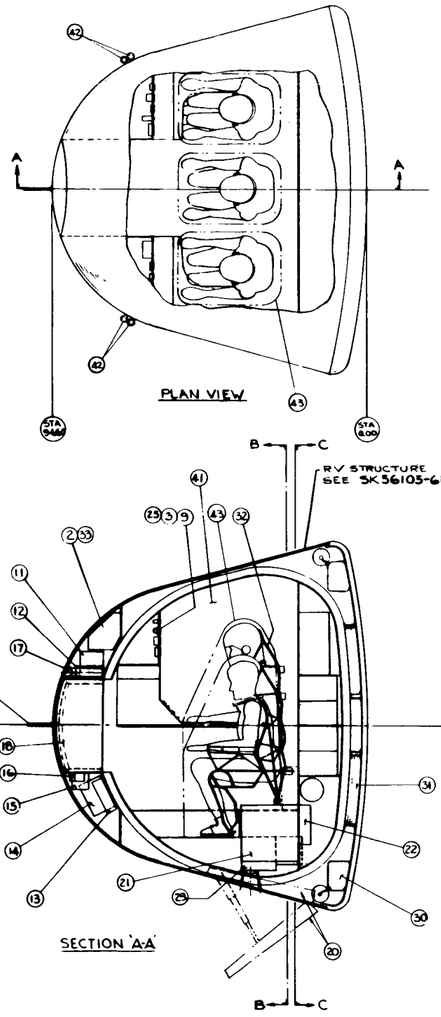
ITEM NO.	
1	POWER SUPPLY
2	TRANSDUCER
3	RCS ENGINES
4	CRAFT
5	OXYGEN SUPPLY
6	DISPOSAL SUPPLY
7	ABORT ROCKETS (SOLID)
8	MID COURSE ENGINE
9	LUNAR ORBIT/RTE ORBIT ENGINE
10	PNEUMATIC TANK (TORUS)
11	FUEL TANK (")
12	OXIDIZER TANK (")
13	ATTITUDE CONTROL STORAGE TANKS
14	FLOTATION BAGS
15	CO_2 REMOVAL UNIT (INC. ODORING & TOXIC GAS REMOVAL UNIT)
16	HEAT EXCHANGER (INC. WATER REMOVAL UNIT)
17	FIRE CONTROL
18	PERSONAL EQUIPMENT
19	RECREATION EQUIPMENT
20	TOILET
21	BATTERY COMPARTMENT
22	WATER TANK
23	STABLE PLATFORM
24	PLATFORM ELECTRONICS
25	COMPUTER
26	GALLEY (INC. FOOD)
27	SURVIVAL EQUIPMENT
28	BEDDING & CLOTHES
29	COMMUNICATIONS EQUIPMENT
30	CRAFT ELECTRIC
31	FLARES
32	LARGE INSTRUMENTATION & CONTROLS
33	MEDIUMSIZE " " "
34	LARGE " " "
35	CONSOLE
36	LIGHTS
37	[illegible]
38	PARACHUTES
39	[illegible]
40	[illegible]
41	[illegible]
42	ANTENNA
43	SPACE SUPERSUIT

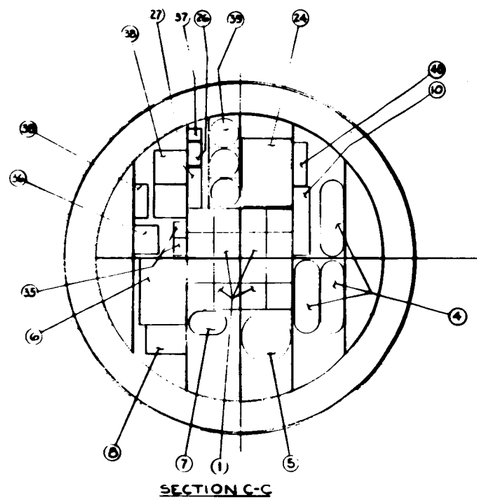
DATA	
[illegible]	
[illegible]	

FOLD OUT #2

Figure XVI-2-2. Inboard profile, C-1 configuration

CONFIDENTIAL





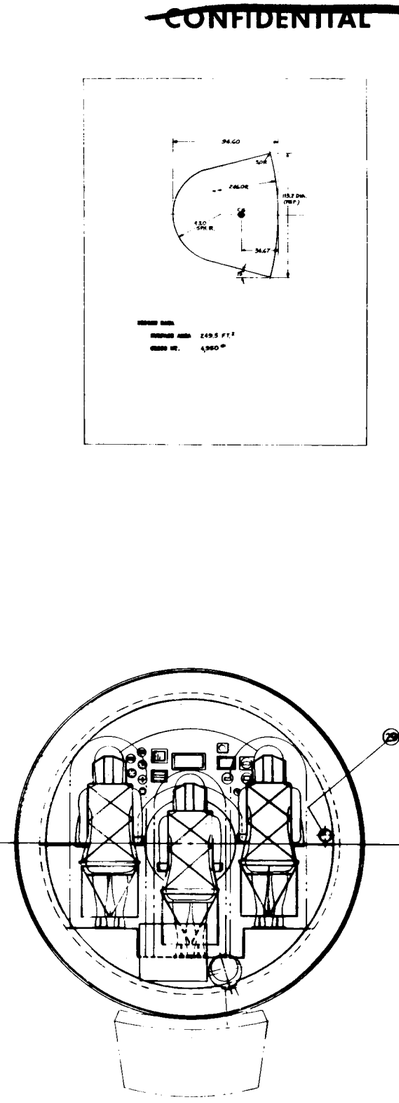


Figure XVI-2-3. Inboard profile, D-2 configuration

pounds, lightest of all re-entry vehicles analyzed. Crew members are seated three abreast, facing aft during re-entry, and landing is implemented by parachutes and impact bag.

SPACE VEHICLE ARRANGEMENT STUDIES FOR SEMI-BALLISTIC RE-ENTRY VEHICLES

General

Arrangement studies for the APOLLO space vehicle have been conducted to develop an over-all configuration and general arrangement of re-entry vehicle, mission area, subsystems, and components which will meet the mission requirements at the lowest practical system weight. In all cases, materials and types of construction considered are in current usage within industry, or are at a development status compatible with the APOLLO delivery schedule. For the B, C, and D R/V shapes previously discussed, both integral and modular arrangements were studied. In all cases, the modular arrangement of a separate mission area resulted in a lighter over-all vehicle weight.

This is due primarily to the larger integrated R/V having both structural and heat protection weight penalties applied to the larger wetted area. For the glide class of vehicles to be described later, only modular arrangements were considered.

This is due primarily to the larger integrated R/V having both structural and heat protection weight penalties applied to the larger wetted area.

In addition to the weight objective, crew space and orientation, abort vehicle stability, and drag requirements during launch and atmospheric abort were vital considerations in over-all vehicle shape and arrangement. Packaging of the R/V as far forward as possible within a bi-conic nose fairing met low drag requirement, with a minimum of afterbody structure required for abort vehicle stability. This arrangement permits location of the mission module, or an airlock, forward of the R/V, and installation of the heavy propulsion system components aft of the R/V.

This arrangement also results in containment of the sub-systems within the lowest wetted area envelope, and corresponding lowest external shell weight.

Meteoroid protection is effectively provided for internal components by utilizing the external shell as a meteoroid "bumper" to shatter moderate size particles penetrating

the external shell. To further minimize the effect of meteoroids, critical components are made redundant in all possible cases, such as use of multiple rocket motors, and sectionalized or modular propellant tanks.

The radiators used for thermal control are integrated into the vehicle shell structure in order to provide the large required radiator areas at the lowest possible weight penalty.

The following discussion describes various total vehicle arrangements studied for each R/V shape.

Integral D Configuration, D-1

The R/V is located forward in the arrangement shown on Figure XVI-2-4a and XVI-2-4b. Note that the R/V is installed nose aft with the crew facing forward. This installation results in the most favorable man orientation to tolerate accelerations during launch, abort, and re-entry. Entrance to the R/V is through an access hatch on the side of the forward nose fairing and a pressure hatch at the R/V aft end. This configuration is 12 feet in diameter at the cylindrical section and 37 feet in length. The entire external shell would be separated for abort in order to provide stability. Abort rockets are located on the aft flare with nozzles canted to direct thrust through the abort-vehicle center of gravity.

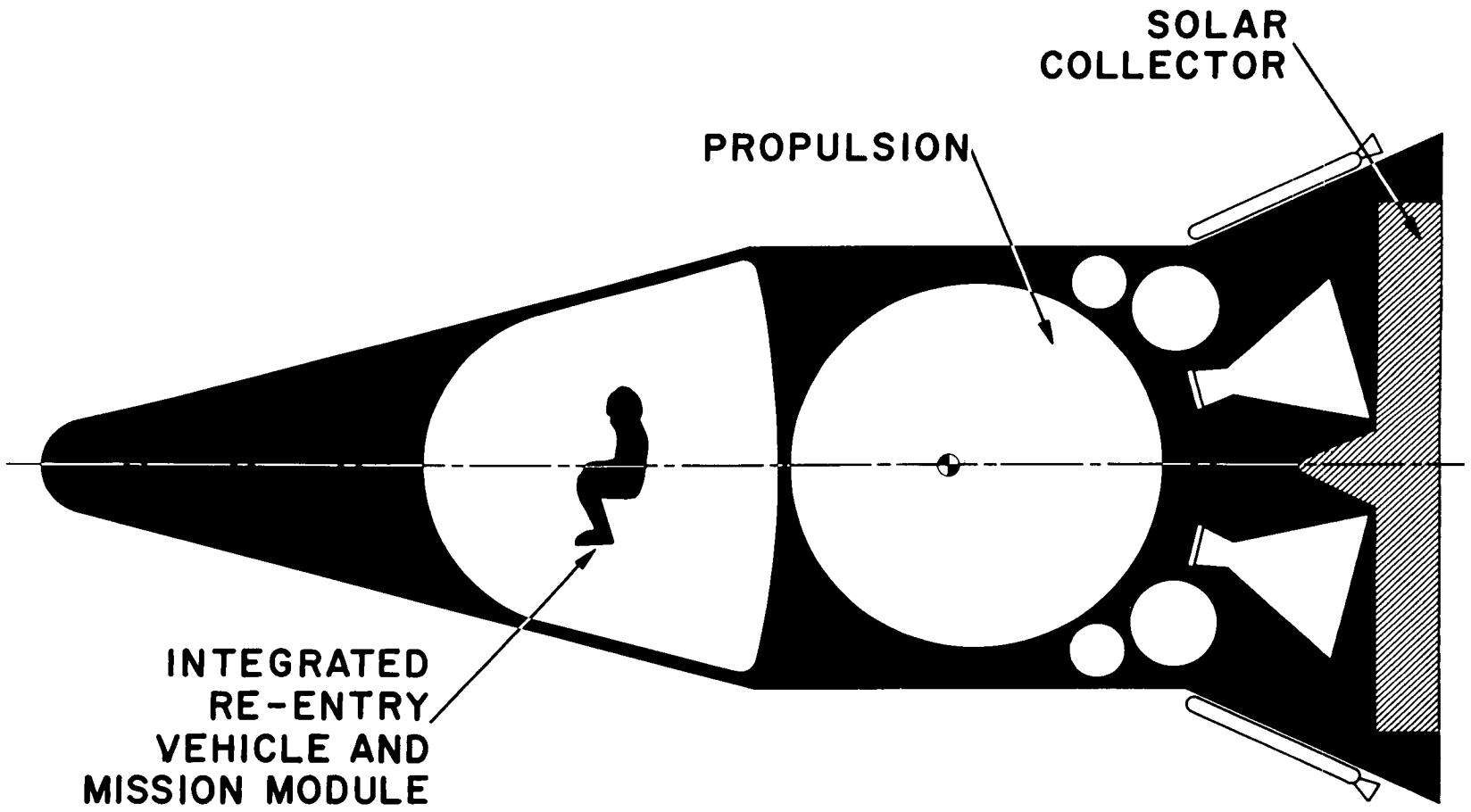
The propulsion system and solar collector are located aft of the R/V, and are left behind on the booster during suborbital abort. In normal operation, the entire vehicle is separated from the booster at the aft interface.

Radiators are installed along the cylindrical external shell section and the solar collector is stored in the vehicle aft flare section. The collector is deployed after vehicle separation from the booster.

Weight of this arrangement is 20,400 pounds.

Modular 'D' Arrangement, D-2

This arrangement, Figures XVI-2-5a and XVI-2-5b utilize the lightest weight of all R/V arrangements studied. The cylindrical section is 10 feet in diameter, versus 12



CONFIDENTIAL

CONFIDENTIAL

Figure XVI-2-4a. D-1 configuration arrangement

Figure XVI-2-4b. D-1 configuration outline

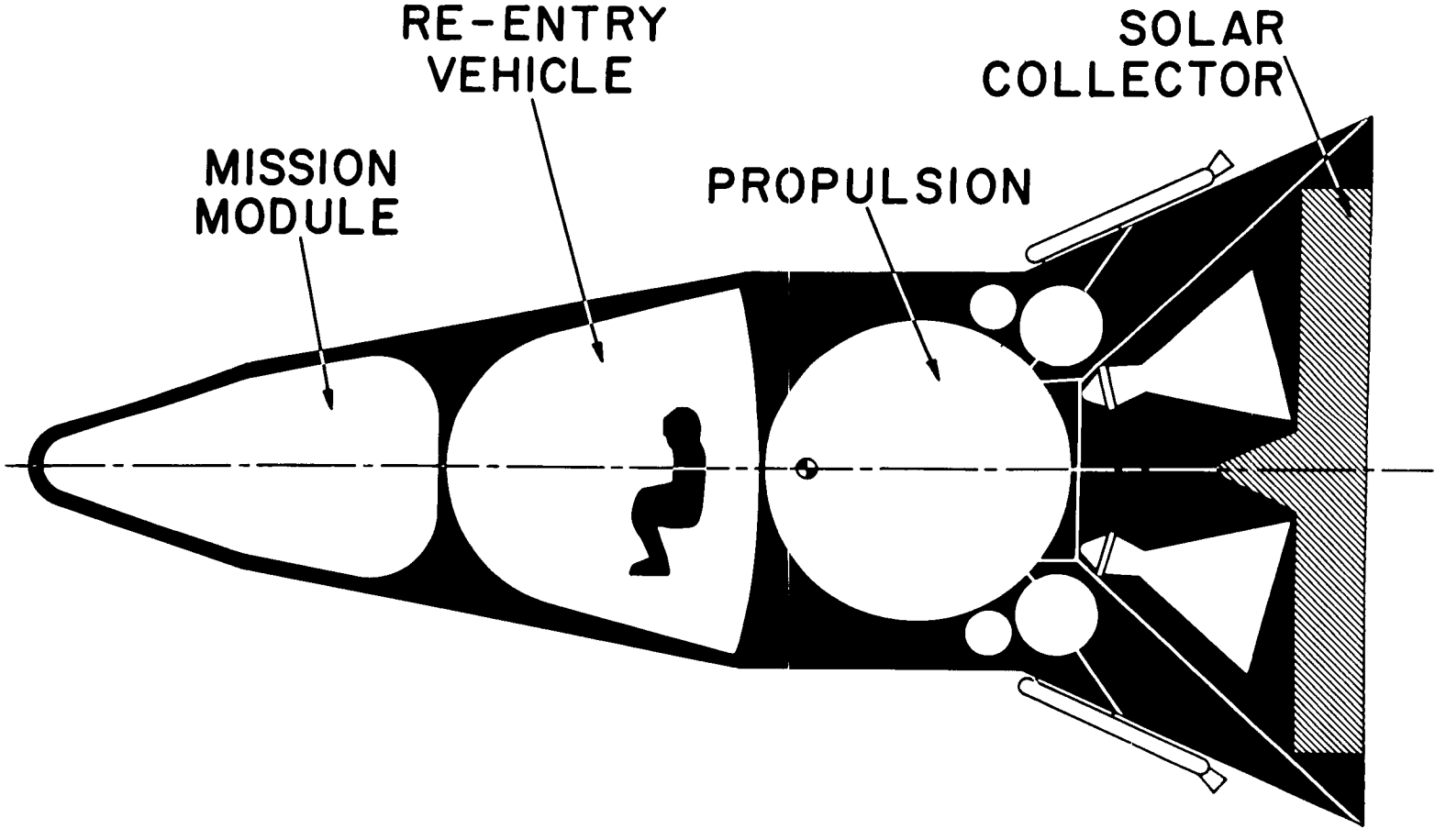


Figure XVI-2-5a. D-2 configuration arrangement

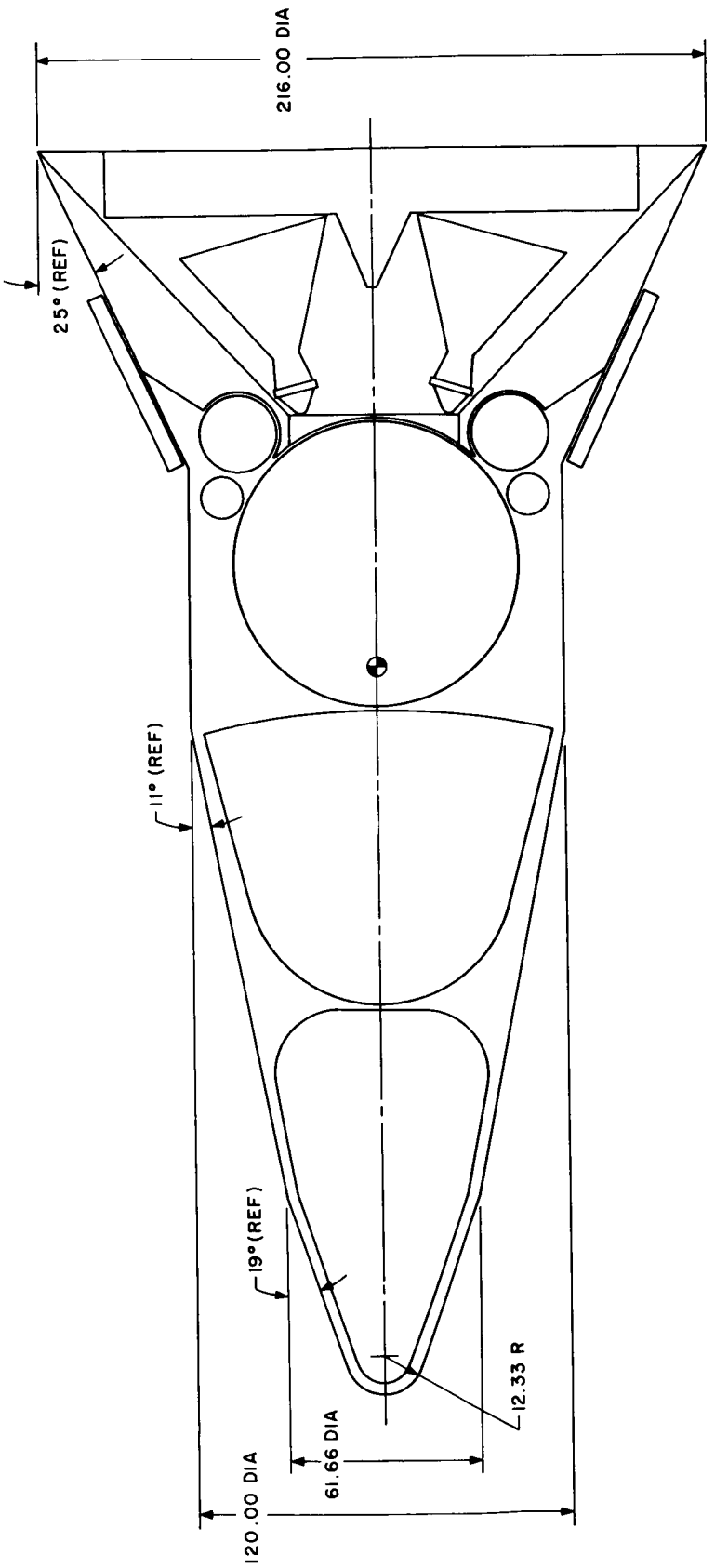


Figure XVI-2-5b. D-2 configuration outline

feet for the integrated version shown before, and over-all length is 33-1/2 feet versus 37 feet. In this arrangement, the mission module, which also serves as an air lock, is located in the forward nose fairing area just forward of the R/V. Entrance is through an access hatch and pressure door in the R/V aft end as before.

The sub-orbital abort separation interface is located forward of the booster interface at a flare diameter of approximately 13 feet, resulting in lower abort vehicle drag due to the reduced vehicle base area.

During sub-orbital abort, the propulsion system and solar collector remain with the booster, and abort rockets are located on the abort vehicle flare section. The R/V is separated from the shell structure by initial structural separation of the shell at the forward juncture of the cylindrical section and subsequent release and ejection of the R/V after small rockets have separated the shell segments. The separation rockets on the nose structure provide positive separation of the nose shell and R/V.

The weight of this arrangement is 16,476 pounds, and a detailed weight breakdown is given at the end of this section.

Integrated 'C' Arrangement,C-1

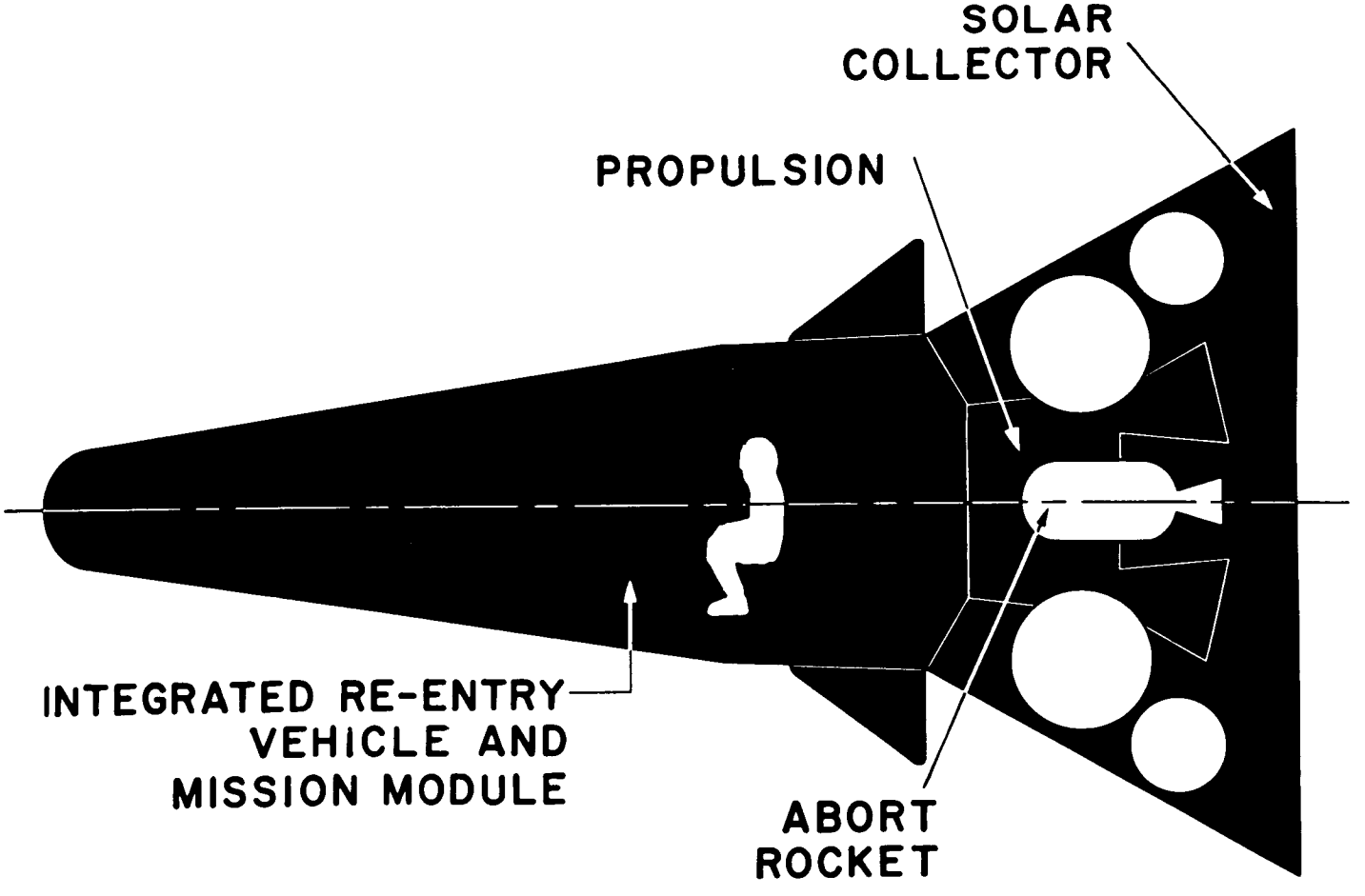
The arrangement shown on Figures XVI-2-6a and XVI-2-6b provide the best solution, weight-wise, for an integral family of "C" R/V configurations, but results is an undesirable man orientation during re-entry and impact (eyeballs out deceleration). In this arrangement the R/V and abort rockets separate at the R/V attach interface.

Weight of this configuration is 24,300 pounds, due primarily to higher heat protection weight. Note that, in this case, the R/V is not protected from meteoroid impacts by the vehicle shell.

Modular 'C' Arrangement, C-2

This arrangement, shown in Figure XVI-2-7, locates the long R/V with nose aft in the space vehicle to achieve the desired crew orientation, and utilizes a mission module in the forward fairing section.

CONFIDENTIAL



CONFIDENTIAL

Figure XVI-2-6a. C-1 configuration arrangement

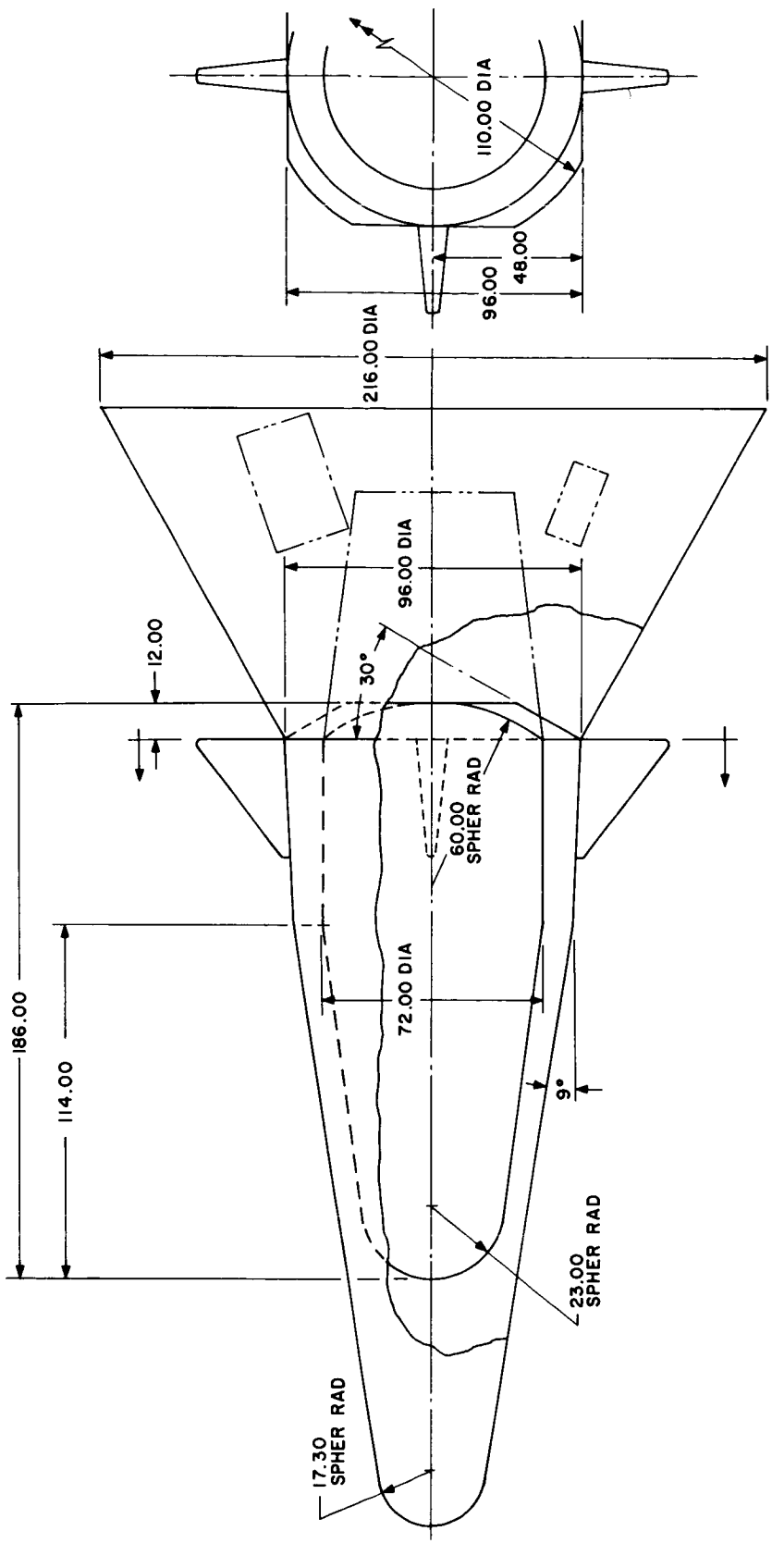


Figure XVI-2-6b. C-1 configuration outline

Figure XVI-2-7. C-2 configuration arrangement

The R/V and entire external shell would be aborted, with propulsion and collector remaining attached to the booster.

Note that for this particular case, abort rockets are shown in a forward location, but could be located on the aft flare section as before.

Weight is 22,000 pounds, and separation and packaging problèms are quite severe.

Integral 'B' Arrangement, B-1

The integrated "B" vehicle shown on Figure XVI-2-8 requires an R/V 13-1/2 feet in diameter and the over-all vehicle is 33-1/2 feet in length. Access is through a hatch and air lock into the R/V aft hatch. On abort, the solar collector and propulsion remain with the booster, as in previous arrangements. One difficulty with this vehicle is maintaining a stable abort vehicle due to the aft location of the large diameter R/V. Weight for this arrangement is 22,700 pounds.

Modular 'B' Arrangement, B-2

The modular B vehicle shown on Figures XVI-2-9a and XVI-2-9b is similar to the D-2 configuration. The mission module is located forward of the R/V in the nose fairing section and propulsion and solar collector are installed in the aft flare section. Multiple torus tanks are shown here to utilize most efficiently the available volume. Abort rockets are shown on the forward section, but could be located on the flare as shown on previous arrangements. The entire 18-foot diameter shell is required for abort stability for the short vehicle shown, but a smaller diameter flare could be used with an additional three to four feet of cylinder.

Increased weight over the modular "D" configuration is due primarily to the heavier R/V and increased propulsion weight.

Over-all length is 26 feet, and total weight is 17,500 pounds.

Alternate Integral 'B' Arrangements

The vehicle shown on Figures XVI-2-10a and XVI-2-10b is approximately 34 feet long and is the full-scale mock-up configuration. Tractor abort rockets are shown, and the

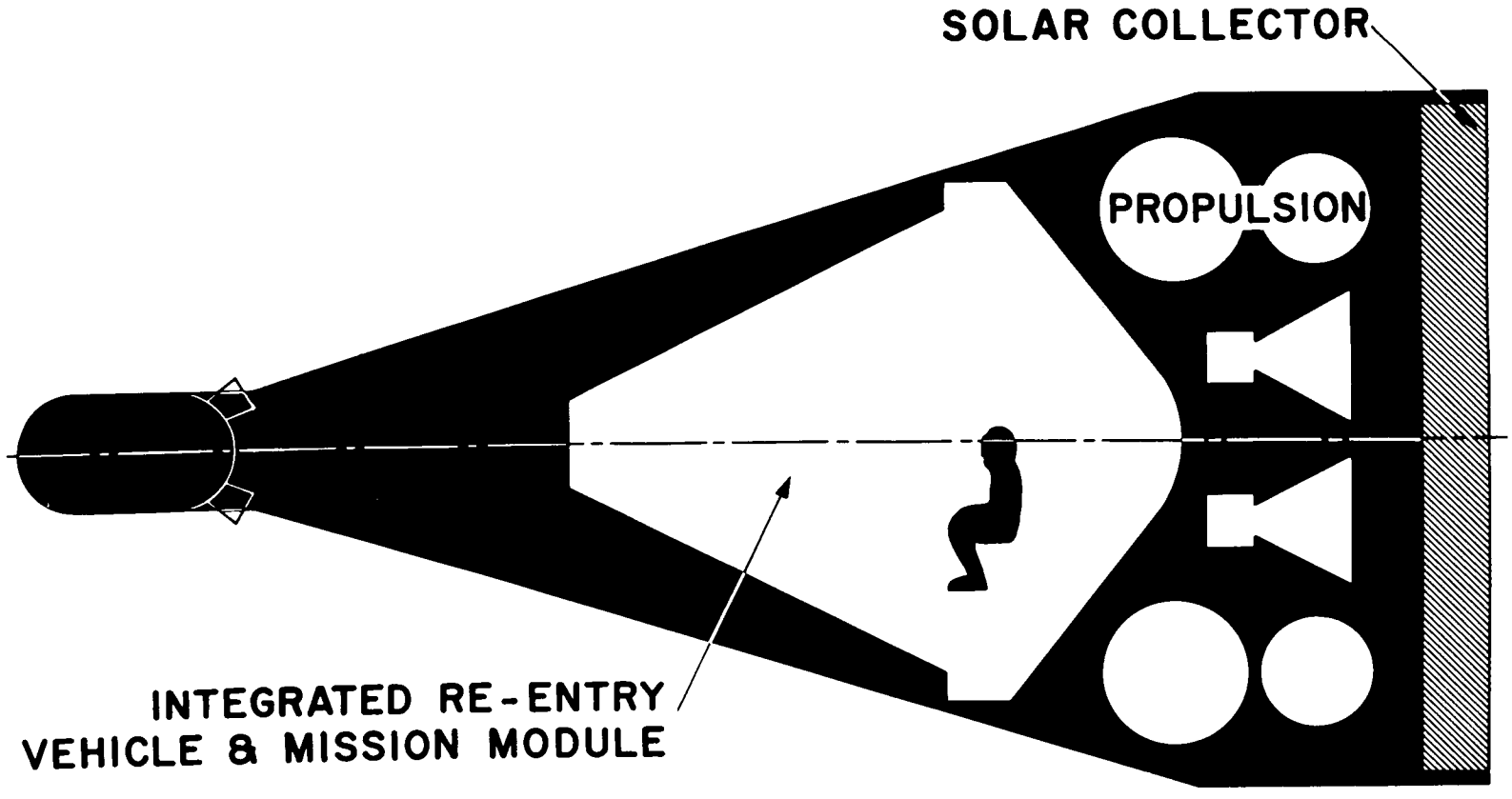


Figure XVI-2-8. B-1 configuration arrangement

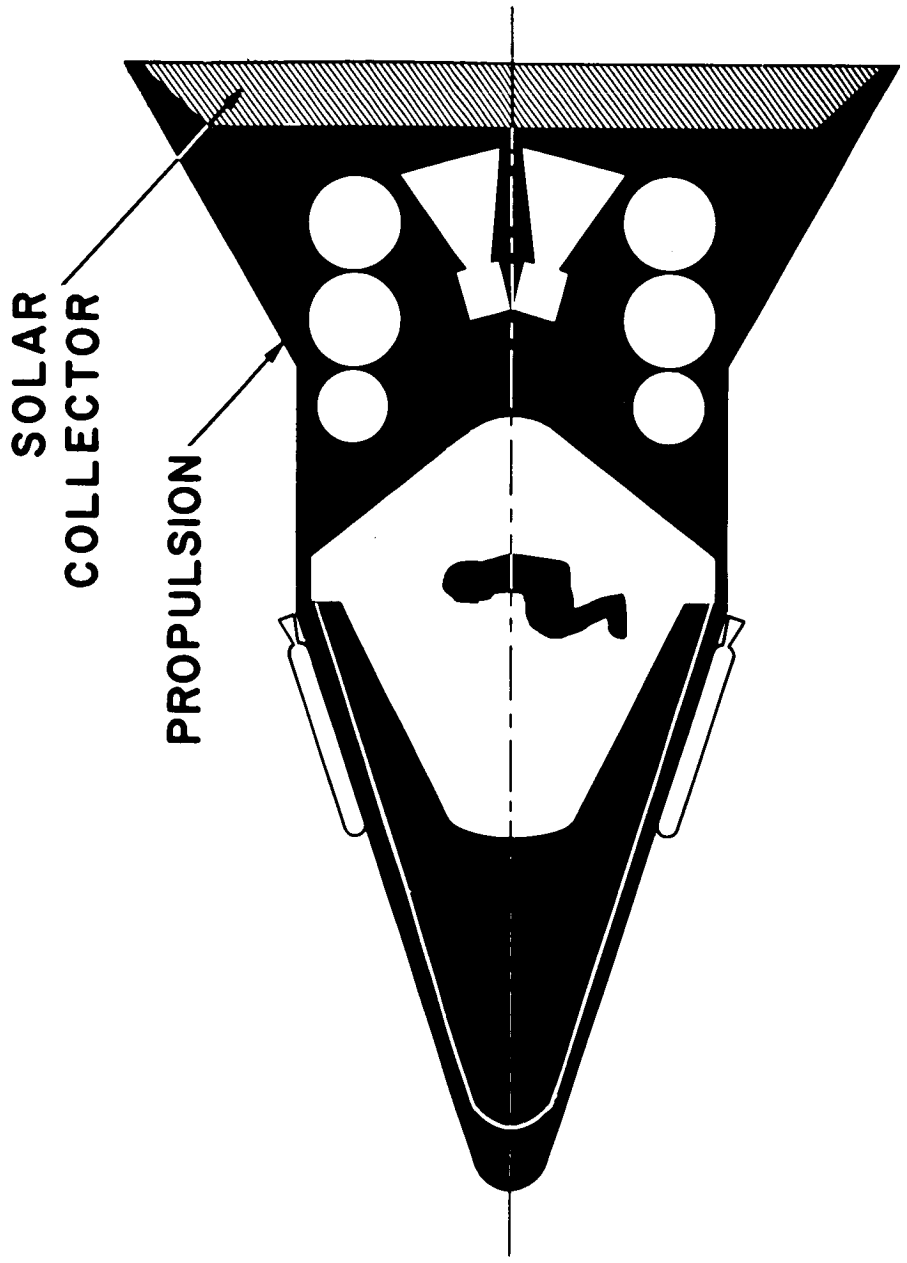


Figure XVI-2-9a. B-2 configuration arrangement

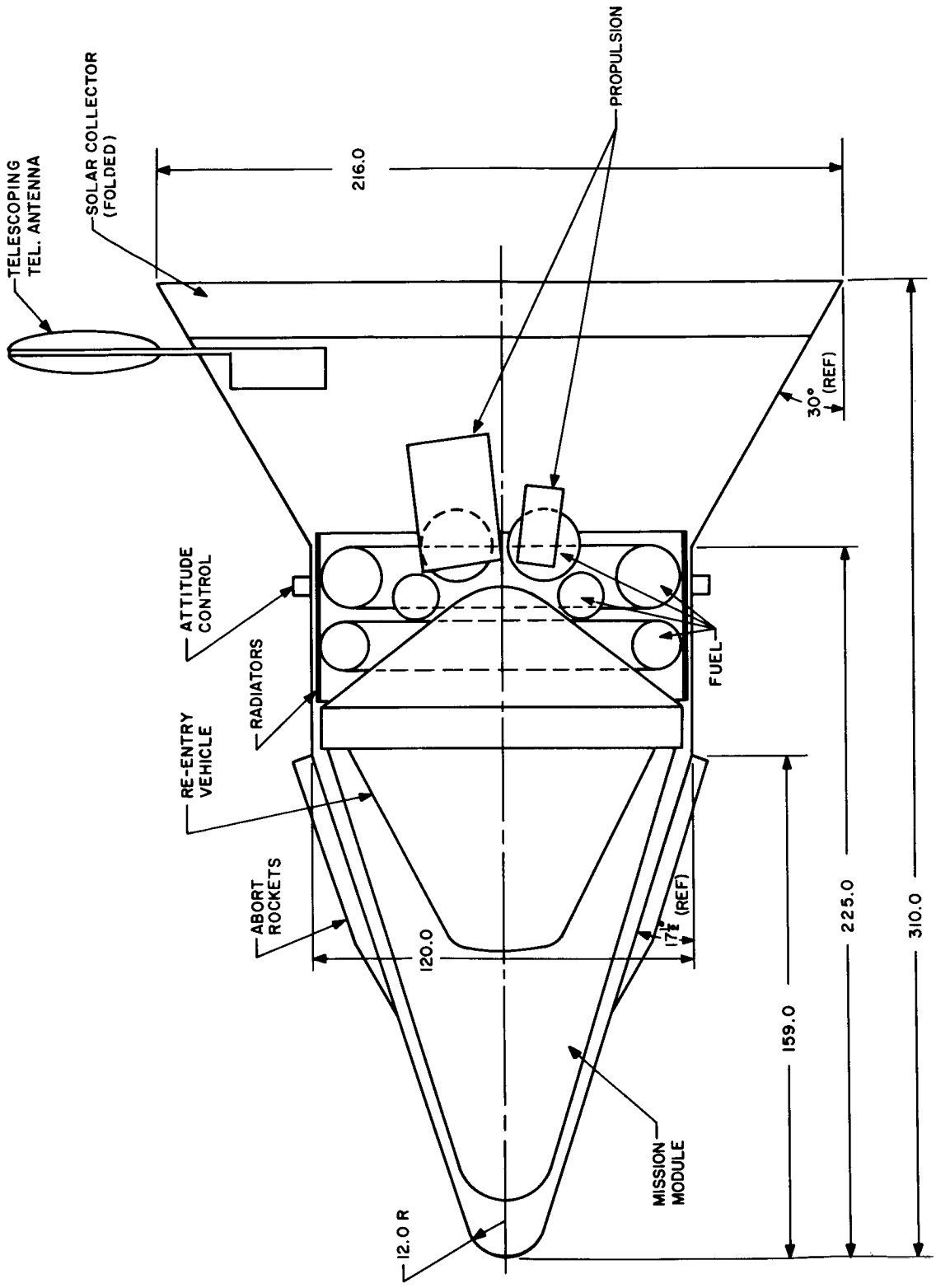
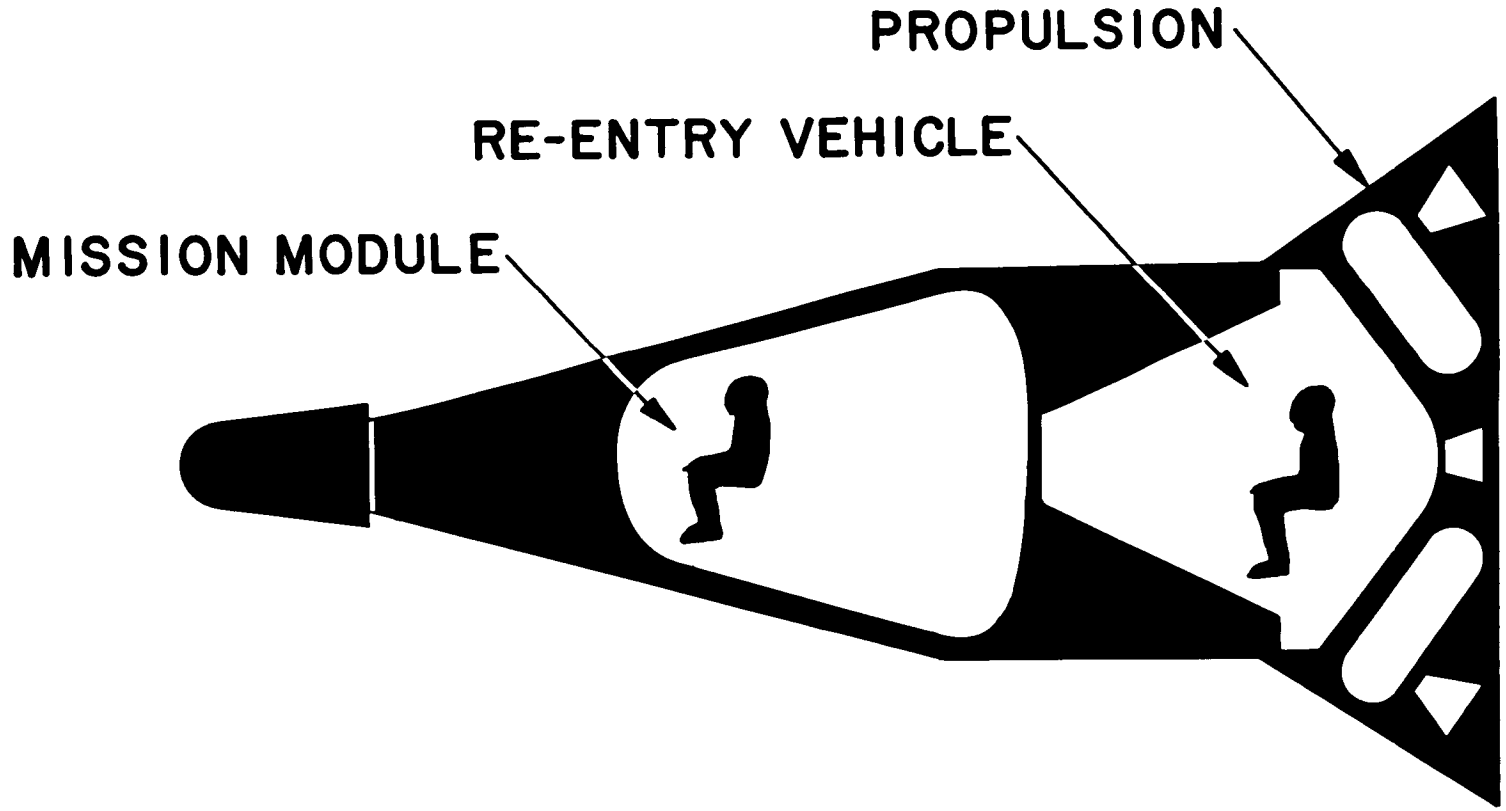


Figure XVI-2-9b. B-2 configuration outline

CONFIDENTIAL



CONFIDENTIAL

Figure XVI-2-10a. B-2 configuration (mock-up) arrangement

CONFIDENTIAL

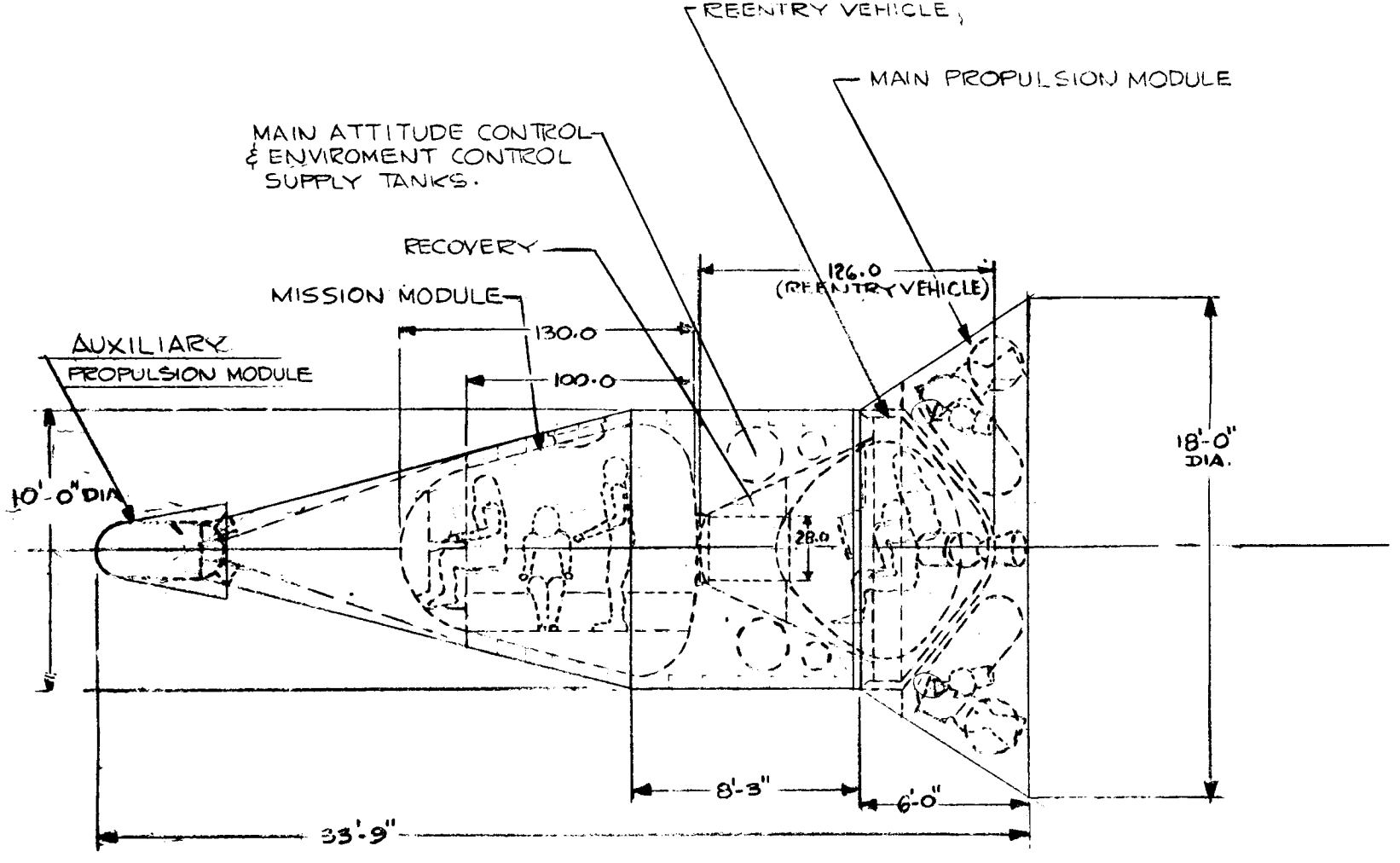


Figure XVI-2-10b. Configuration B-2 (mock-up) outline

CONFIDENTIAL

entire vehicle shell is aborted. Weight is approximately 18,000 pounds due to the larger vehicle wetted area.

Figure XVI-2-11 shows a torroidal mission module, and large external fairing. The solar collector is mounted forward of the R/V. Weight for this vehicle is in excess of 20,000 pounds, and the vehicle has high drag.

The arrangement shown on Figures XVI-2-12a and b places the mission module aft of the R/V and requires a hatchway through the R/V forward shield for entrance. The weight is more than 20,000 pounds, and again, high abort drag precludes consideration of this arrangement. The hatch through the forward shield would be made in a manner similar to that of the NERV installation; however, the installation would be relatively heavy and complex due to the impact bag installation and shield extension requirement prior to landing. From study on such an arrangement, it is concluded that juncture between the R/V and mission module should be at the R/V aft end in all cases.

Figures XVI-2-13 and XVI-2-14, show two additional heavier "B" arrangements studied.

Analyses of all of the arrangements provided the necessary trade-off data to permit a choice of the selected modular arrangement.

INTEGRAL VERSUS MODULAR WEIGHT COMPARISON

Table XVI-2-1 summarizes weights for the best arrangement for each integrated or modular R/V concept. Factors considered in addition to total weight are abort weight and stability requirements, crew orientation, powered flight drag, and meteoroid protection. The large size of the R/V's for the integrated designs contributes most heavily to the higher weights shown.

It is concluded from this study that the modular arrangement best meets the over-all APOLLO requirements and the "D" R/V configuration is the most efficient semi-ballistic re-entry vehicle.

CONFIDENTIAL

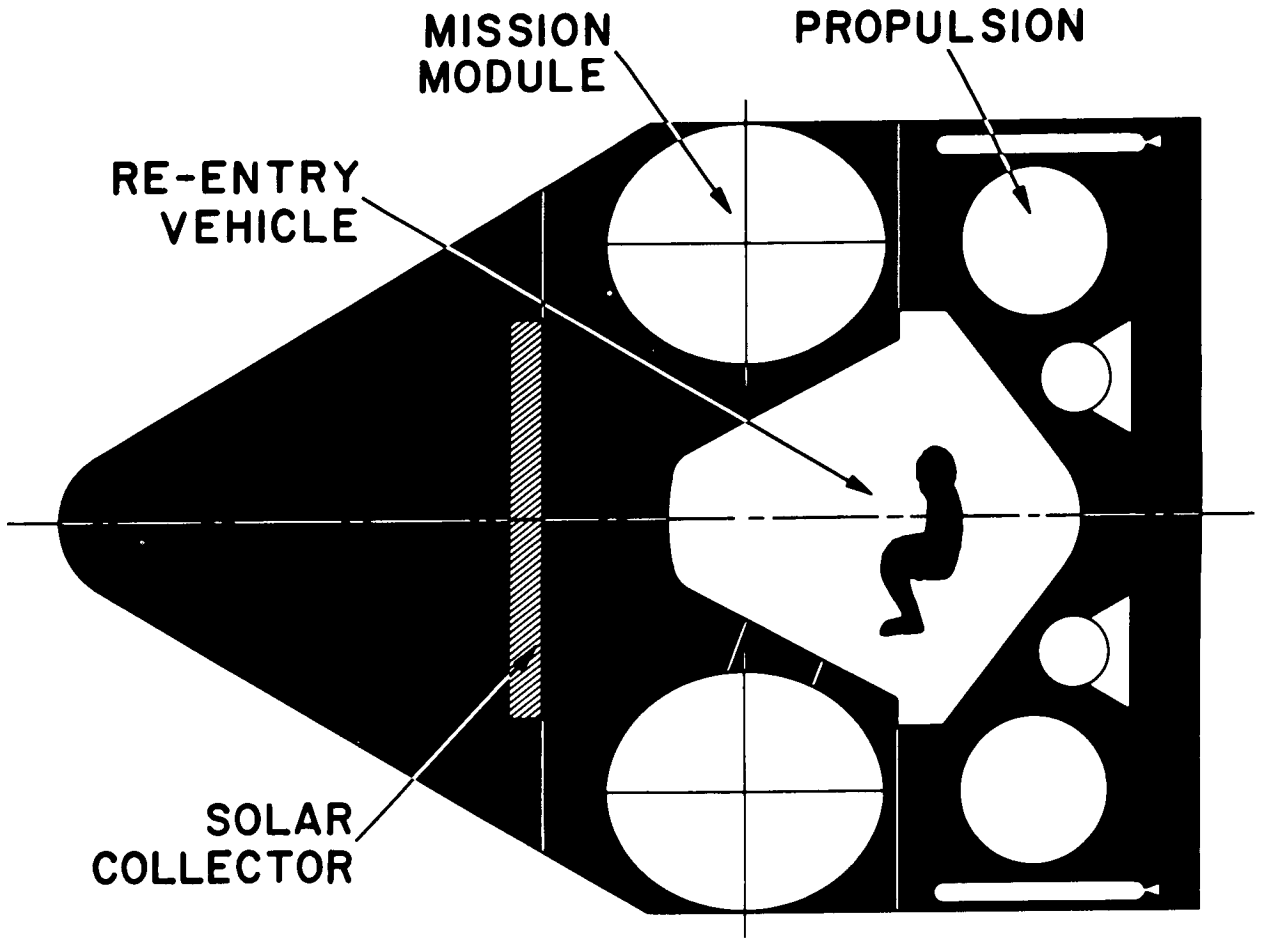


Figure XVI-2-11a. Alternate B-2 configuration arrangement

CONFIDENTIAL

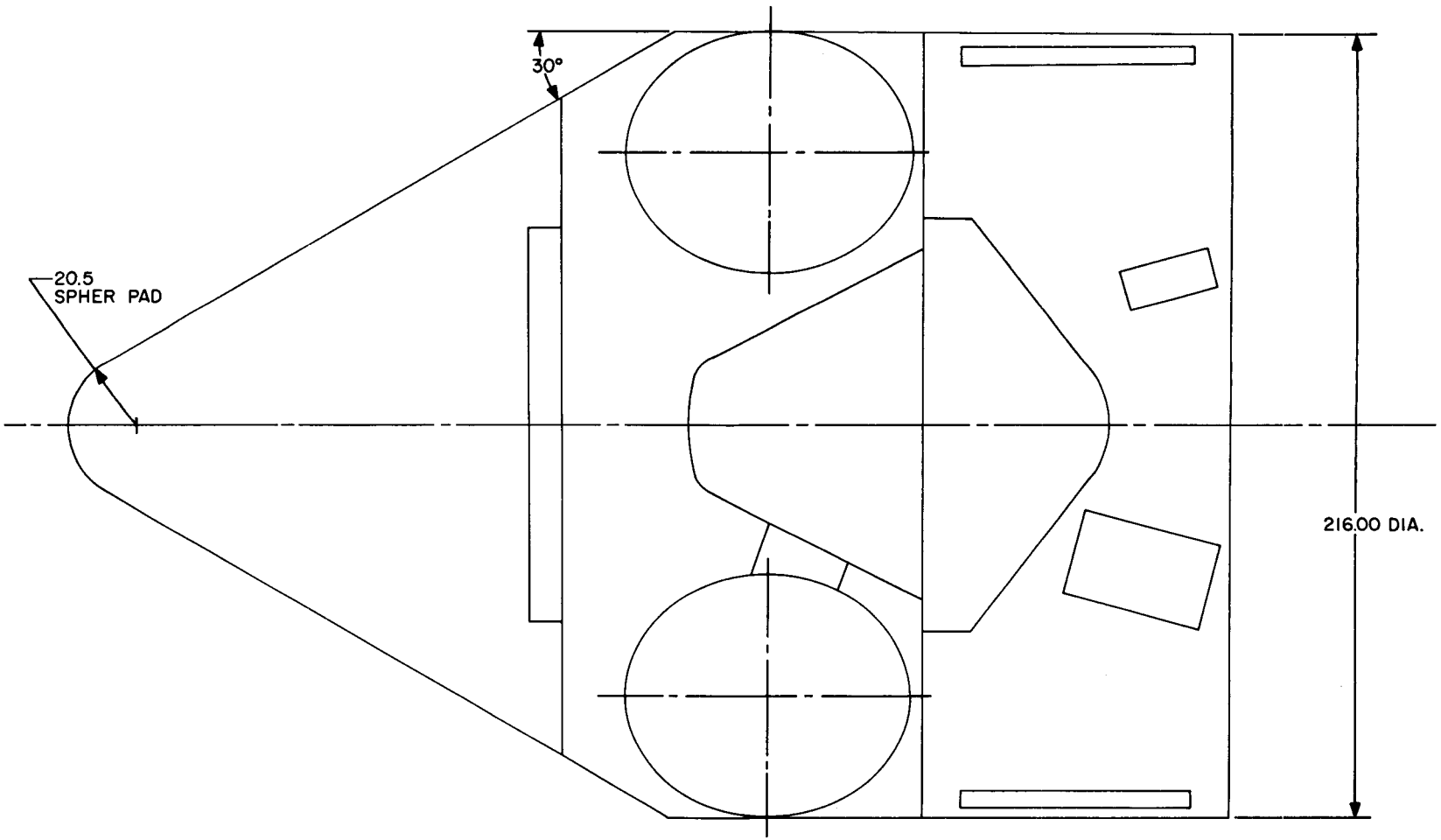


Figure XVI-2-11b. Alternate B-2 configuration outline

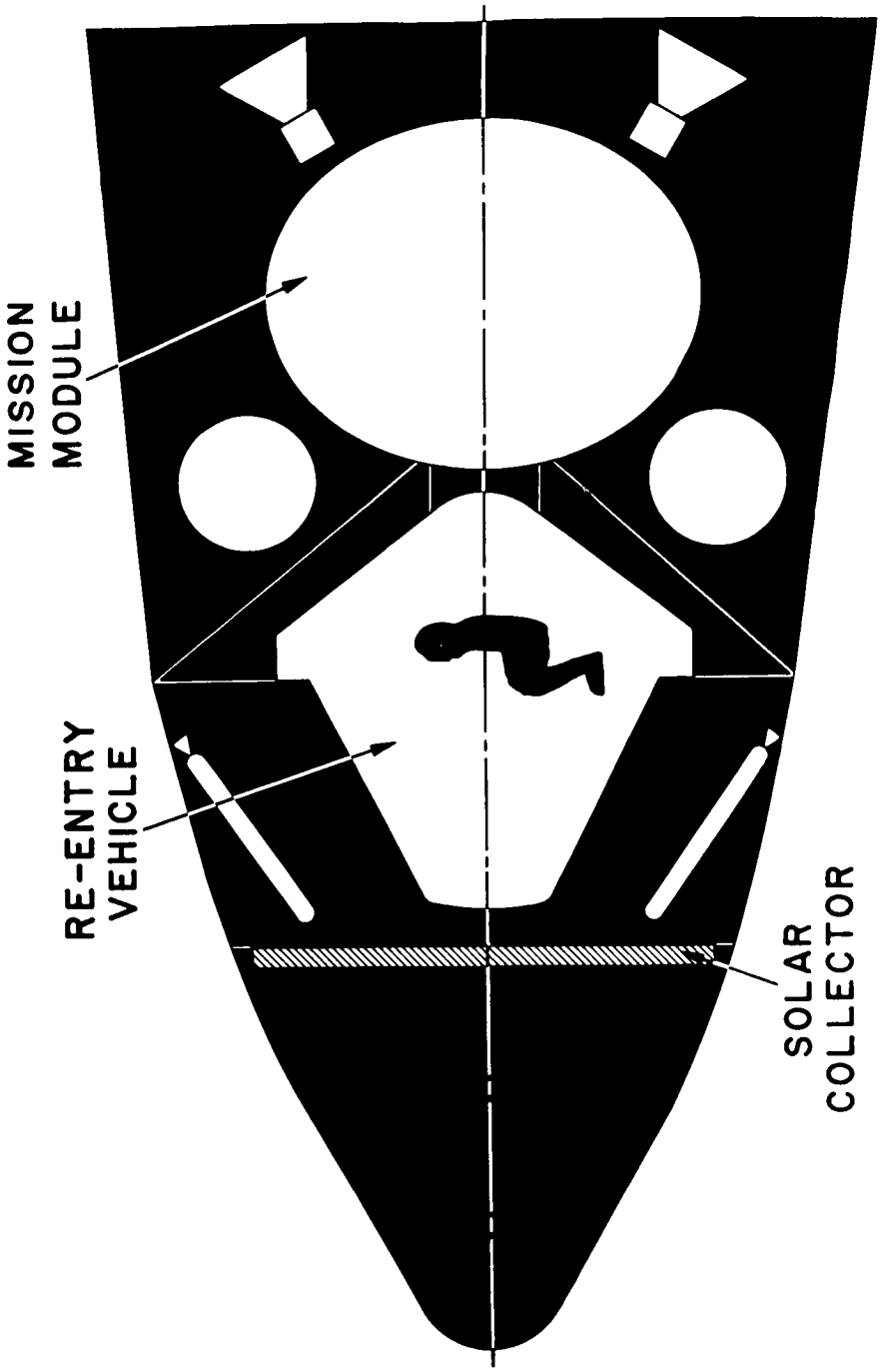
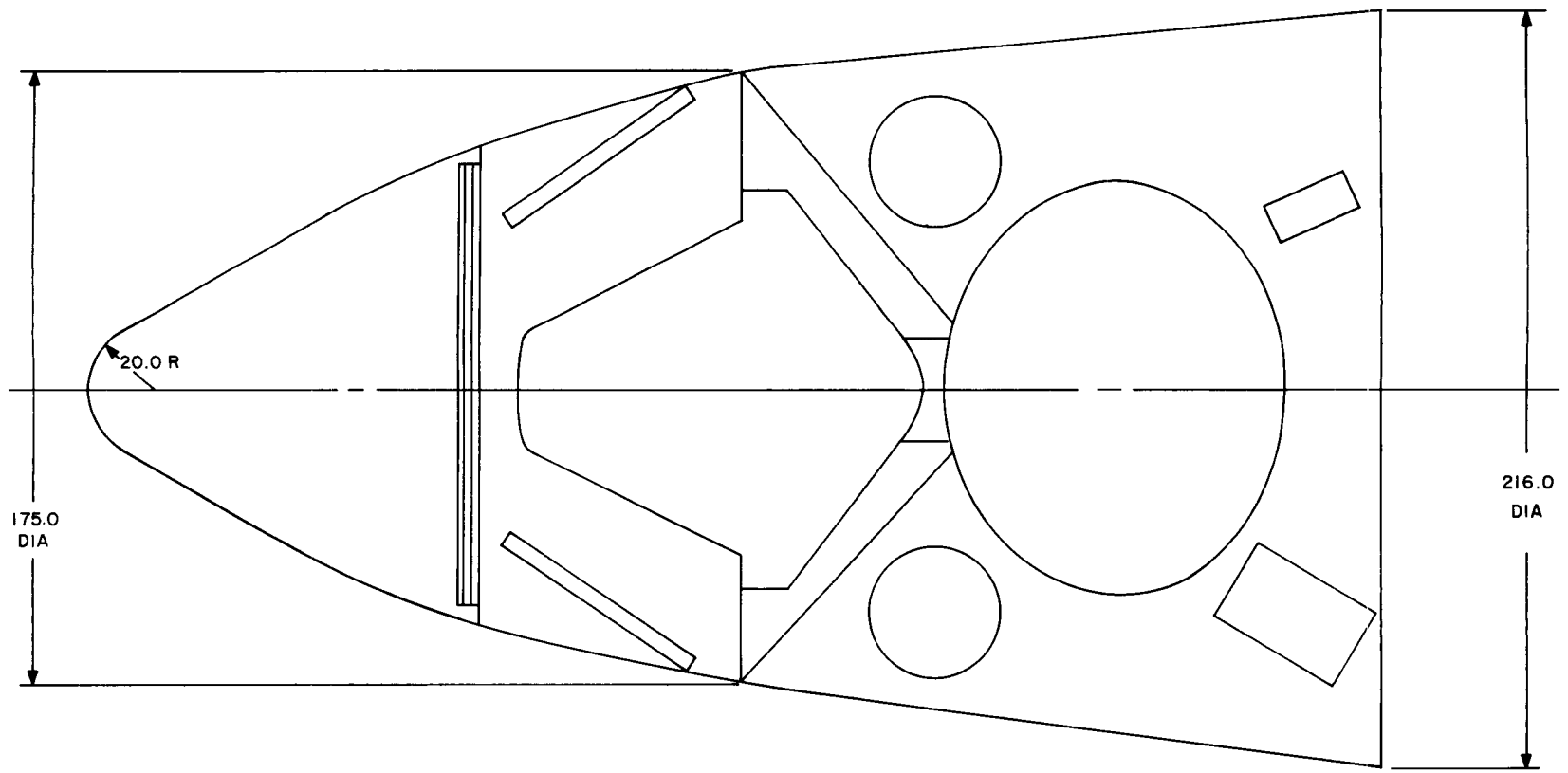


Figure XVI-2-12a. General arrangement, alternate B-2 configuration

CONFIDENTIAL



CONFIDENTIAL

Figure XVI-2-12b. General arrangement outline, alternate B-2 configuration

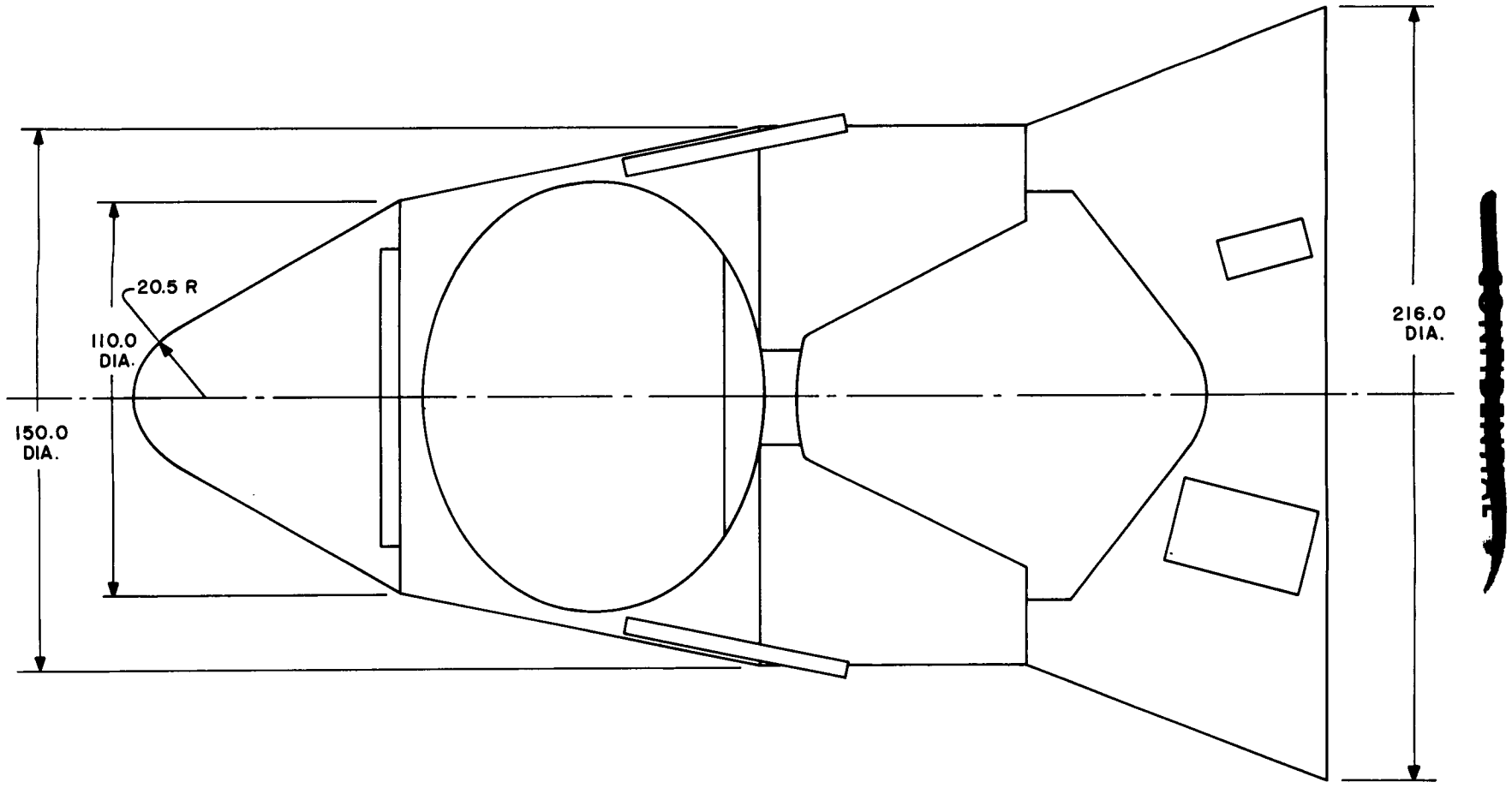


Figure XVI-2-13. General arrangement outline, alternate B-2 configuration

CONFIDENTIAL

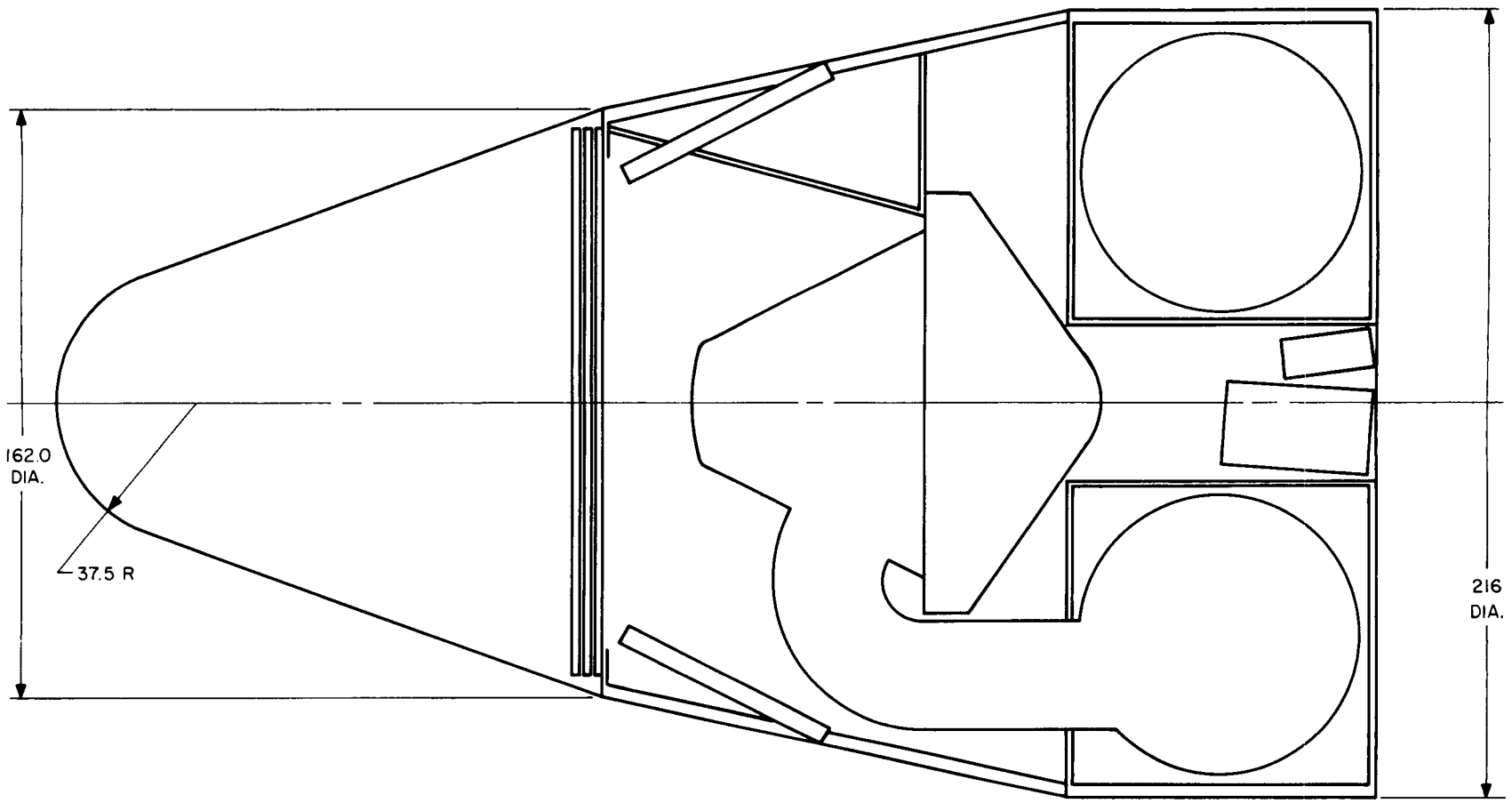


Figure XVI-2-14. General arrangement outline, alternate B-2 configuration

CONFIDENTIAL

TABLE XVI-2-1. INTEGRAL VS MODULAR DESIGN

Weight Summary (Present design capability)

Configuration	Integral Design	Modular Design
B	22, 700	17, 500
C	24, 300	22, 200
D	20, 400	16, 476

Other Considerations:

1. Crew Orientation
2. Abort Stability and Drag
3. Packaging Volume
4. Minimum Abort Weight
5. Rendezvous and Lunar Landing Versatility

CONCLUSIONS OF SEMI-BALLISTIC VEHICLE STUDIES

Significant conclusions drawn from these studies are summarized as follows:

(1) Crew Orientation. Installation of the R/V nose-down on launch, with the men facing forward (facing aft in the R/V), results in an "eyeballs in" crew position for launch, abort, and re-entry high-g inertia conditions. This position is considered most desirable for both comfort and maximum effectiveness of the crew. Installation of the mission module forward of the R/V permits orbital operations such as rendezvous, and simplifies access and sealing between the R/V and mission module (the R/V forward heat shield is not pierced).

(2) The cone cylinder flare space vehicle shape shown for the D general arrangement has the lowest surface area and correspondingly lowest weight of all S/V shapes studied. The ogive shapes were studied to evaluate packaging and arrangement concepts but, in addition to higher weight, were not aerodynamically stable as abort vehicles. The D and B complete vehicle shape is stable during abort, and also has lower bending and shear loads at the S/V-Saturn interface. In order to reduce drag on the abort vehicle, the flare diameter at the abort separation interface is kept to a minimum consistent with stability margin requirements.

(3) A primary design objective has been to achieve the lowest gross weight vehicle consistent with mission requirements and safety. Aft arrangements of the propulsion system and collector permits efficient separation of these items from the remainder of the external S/V shell. For abort, propulsion and collector remain with the booster. This concept results in a light-weight abort configuration, consisting of the R/V and mission module components and utilizes the forward section of the aft flare to assure stability.

(4) The external shell structure provides an effective meteoroid bumper to virtually eliminate meteoroid damage to critical internal areas. In the R/V pressure capsule area the R/V heat shield and sub-structure provide additional shield material to minimize the probability of punctures. Propulsion fuel is packaged in multiple tanks and redundant rocket motors are provided to decrease the over-all effects of meteoritic impact.

(5) The large radiators required for thermal control are integrated into the structural shell to serve as primary load-carrying structure, resulting in maximum utility of the radiator materials and minimum weight design.

(6) "D" vehicle provides the most usable crew space.

SELECTED SEMI-BALLISTIC CONFIGURATION

This section describes in detail the features of the modular "D" configuration, which has been selected as the best overall semi-ballistic vehicle.

Figures XVI-2-15a and b show the placement of primary sub-systems and components in both the R/V and space vehicle. The mission module is located in the nose fairing forward of the R/V. The propulsion and solar collector are installed aft of the R/V in the cylinder and flare section. Multiple propulsion fuel tanks will be provided rather than the large spherical tank shown, and the torus tanks are compartmented to alleviate meteoroid damage.

The abort separation interface is located at approximately one-third of the flare length, and propulsion and the collector remain with the booster on abort.

Actuation of the abort system provides abort vehicle separation and abort rocket ignition milliseconds later. Abort vehicle separation is accomplished by a shaped charge which severs the external shell circumferentially at vehicle station 334. Two seconds after abort rocket burnout, the R/V separation is accomplished by shaped charge separation of the vehicle outer shell at vehicle station 298 and shaped charge separation of the R/V support cylinder. This vehicle separation breaks a connection that ignites separation rockets which propel both forward and aft sections of the external shell away from the R/V. The recovery chute sequence is then initiated at apogee and normal recovery sequences follow.

Normal vehicle separation occurs at the booster interface. After separation of the Space Vehicle (S/V), and booster, the solar collector, and communication antenna are extended and deployed and normal mission sequences initiated. Re-entry vehicle

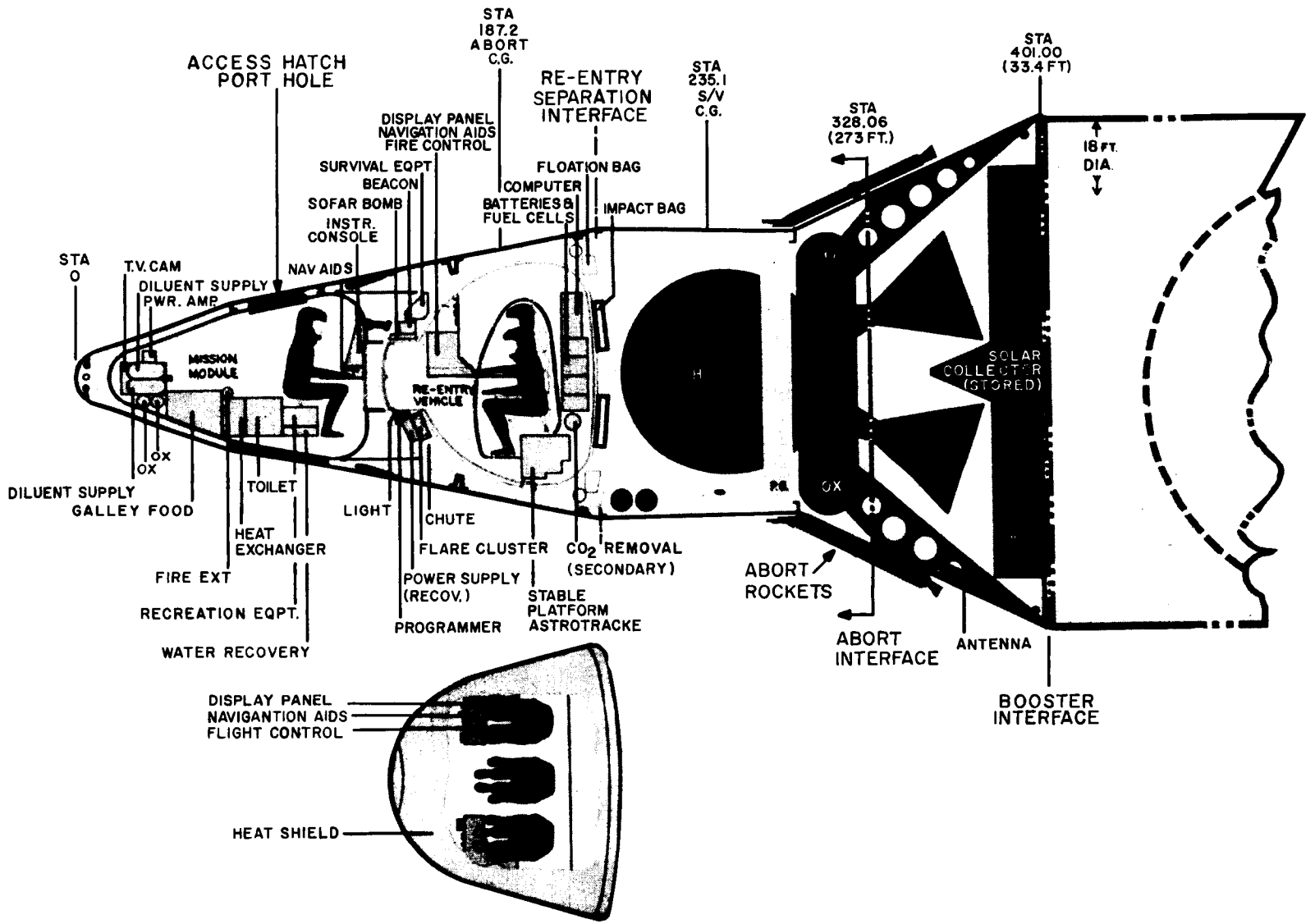
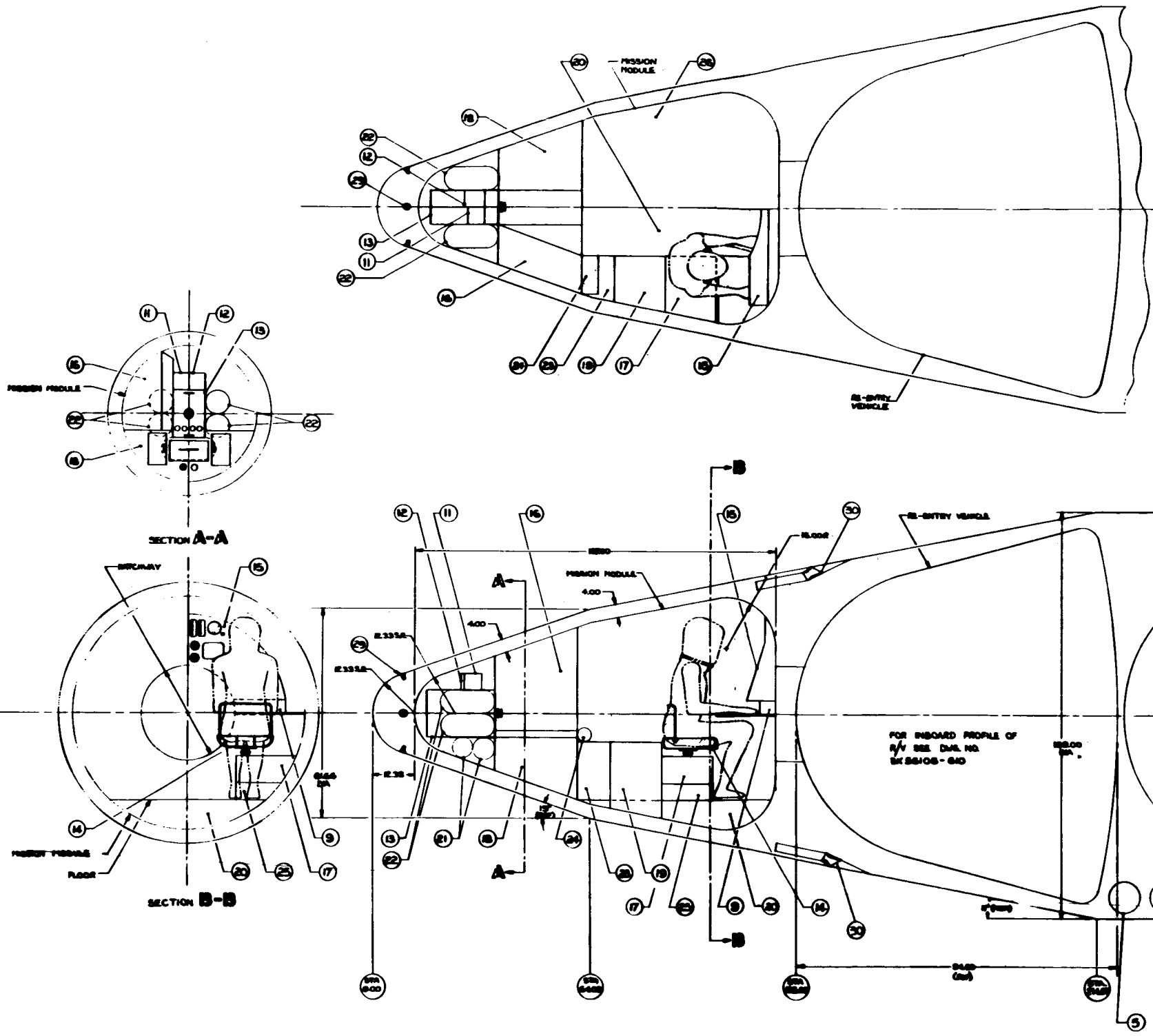
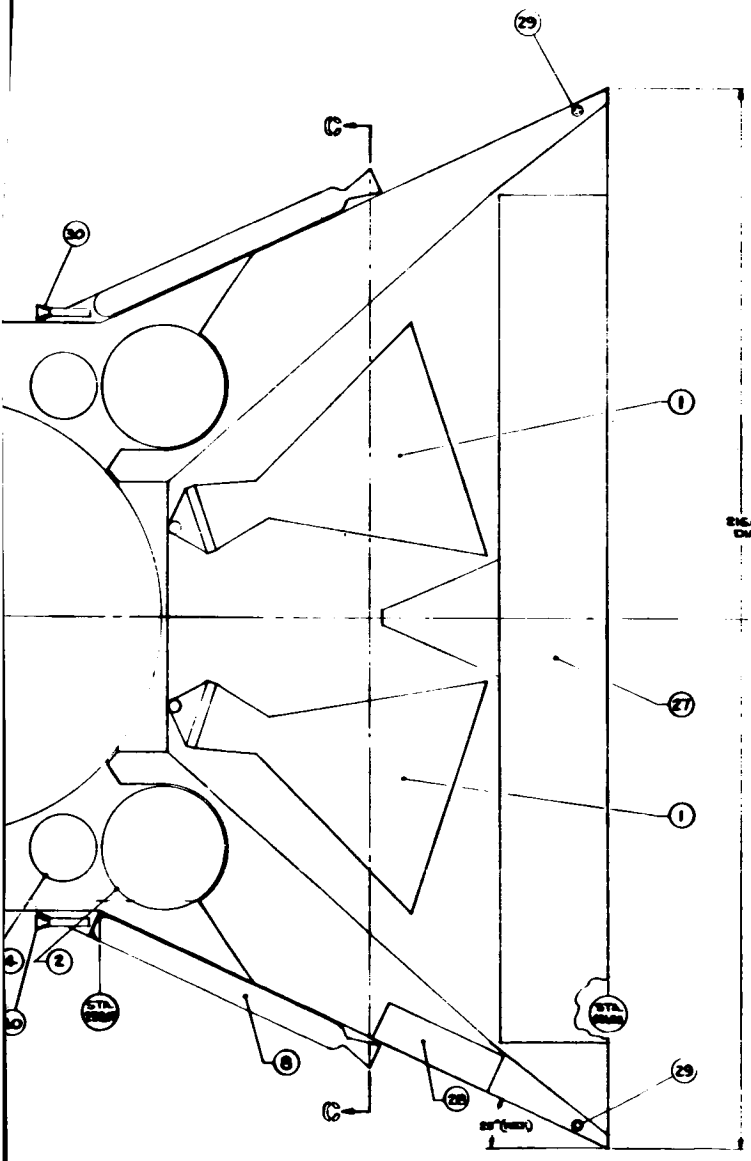


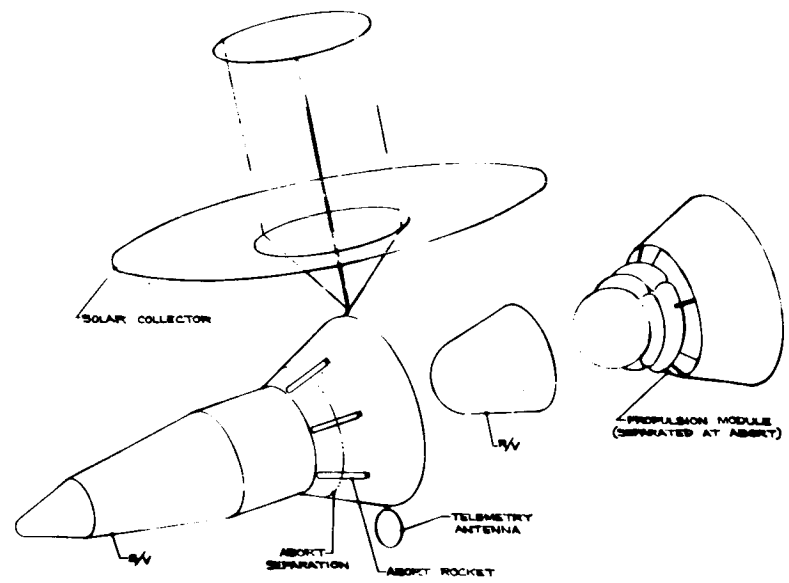
Figure XVI-2-15a. Selected D-2 configuration arrangement

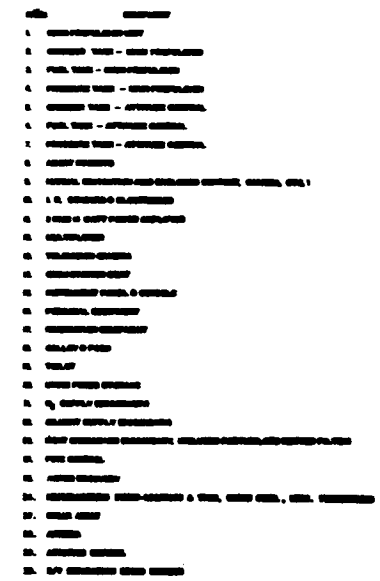
CONFIDENTIAL


MISSION MODULE
RE-ENTRY VEHICLE
SECTION A-A
SECTION B-B
AIRWAY
FLOOR
FOR INBOARD PROFILE OF S/V SEE DWG. NO. SK 58108-640

FOLD-OUT #1







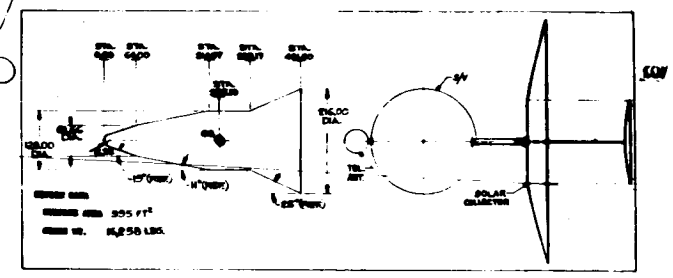


Figure XVI-2-15b. Selected D-2 configuration outline

FOLD OUT #2

separation prior to re-entry is accomplished as previously described for the abort condition. The following major components are located in the re-entry vehicle cabin: (Refer to Figure XVI-2-3)

1. Astrotracker and inertial platform
2. Computer
3. Primary internal environment system, i.e. O_2 supply, CO_2 removal, N and T and H_2O removal
4. Fuel cells and emergency batteries
5. Crew seats and restraining system
6. Secondary pressure system

Secondary pressure systems are inflatable nylon capsules that enclose both the individual seats, crew members, and emergency controls. These capsules have sufficient volume to enable a crew member to don a minimum space suit while inside, in order to make any necessary repairs.

Heavy equipment, such as batteries and the environmental control system, is installed behind the men in order to achieve an adequate center-of-gravity location for re-entry stability. The recovery system is installed between the heat shield and pressure capsule in the aft section. A section of the shield and sub-structure is extended after re-entry to permit initiation of recovery sequences.

All equipment mounted in the R/V and mission module may be readily removed for access to the pressure capsule surface to permit any necessary repairs.

An astrodome is installed on the space vehicle nose, and a port-hole is provided on the external entrance hatch. Access to the R/V is through a pressure door between the R/V and mission module. Pressure equalizing systems are provided to permit opening the doors in the event pressurization is lost in one of the compartments. The mission module thus has the potential to serve as an airlock for orbital operations in rendezvous or extravehicular experiments.

Visual observation from the R/V is provided during re-entry by a periscope, and a window may be installed for use after re-entry if required. The window would be of double wall construction attached to both the pressure capsule and shield support structure. The window would be protected during re-entry by a door which would be jettisoned during parachute deployment.

The present design was based on a Saturn interface of 18' as specified in NASA guidelines. The arrangement selected has some flexibility to accommodate other booster interface diameters, if required.

STRUCTURAL DESCRIPTION AND DESIGN CRITERIA

This section describes the structure, materials, critical loading conditions, and design criteria, for the modular "D" configuration.

Structural Description

The space vehicle and re-entry vehicle primary structure is shown on Figure XVI-2-16, XVI-2-17, and XVI-2-18. External S/V shell structure is of aluminum honeycomb construction. The internal S/V structure consists of the following primary members:

1. Mission module - welded pressure vessel
2. Re-entry vehicle attach cylinder
3. R/V lateral support frames
4. Propulsion support frames (6) similar to water-tower supports

Re-entry vehicle structure, Figure XVI-2-18, consists of an inner pressure capsule, outer sub-structure supporting the heat shield, and intermediate structural frames. Conventional aluminum alloys are proposed for both the S/V and R/V primary structure. Bonded honeycomb core, 2024 aluminum faced sandwich is used for the S/V external shell and formed or machined 2024 or 2014 bulkheads, webs, and keels are used for internal structure. Magnesium alloys will be used where riveted or bolted connections may be employed to take full advantage of the potential weight savings

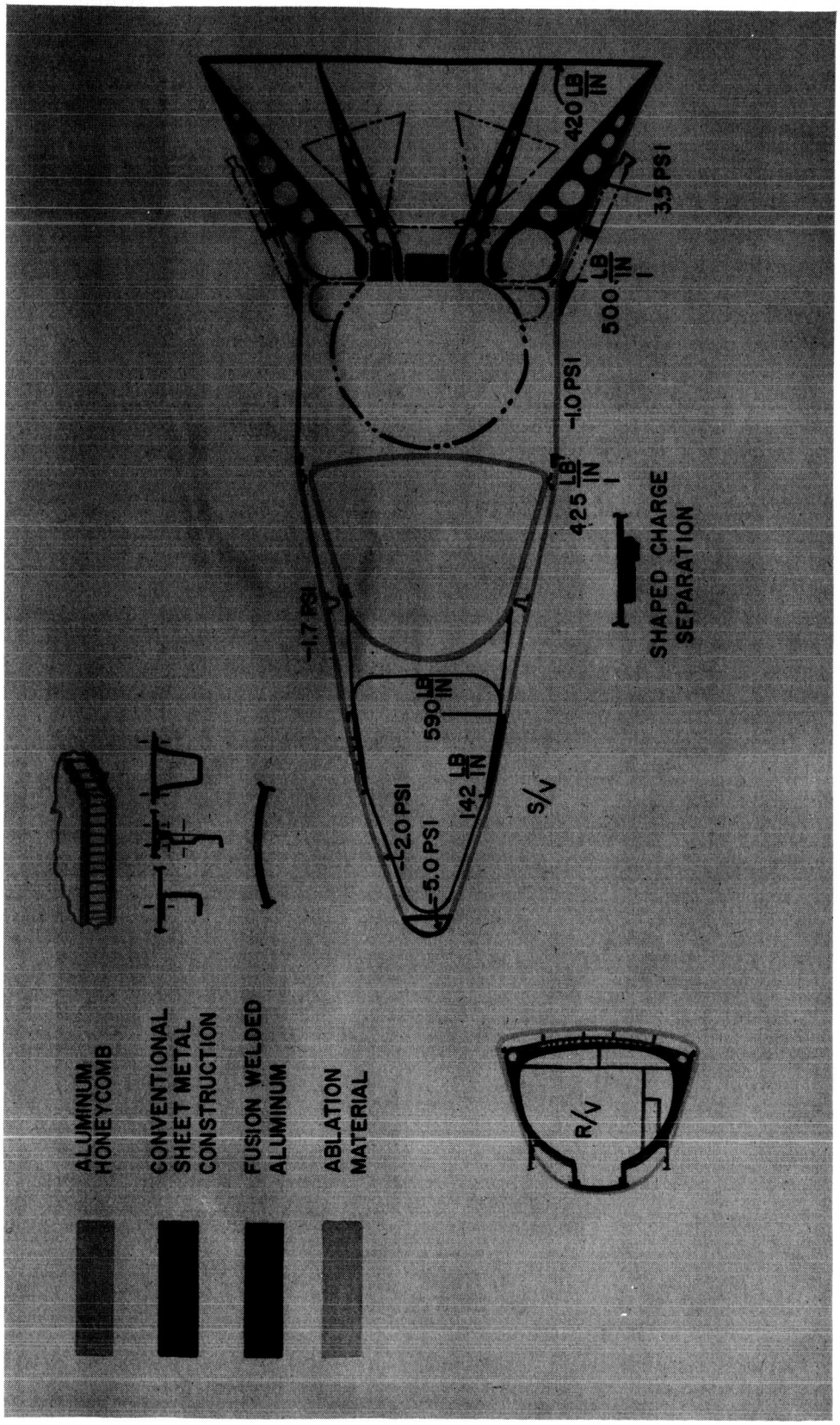
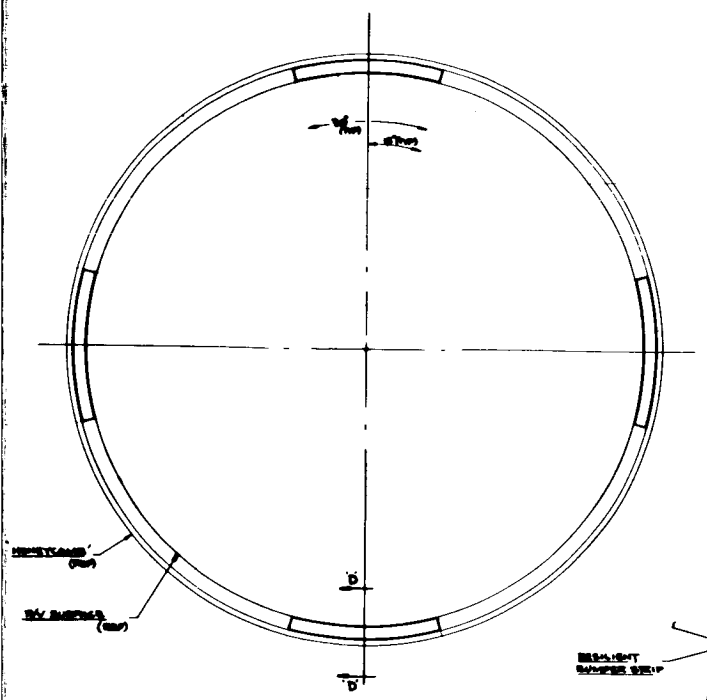
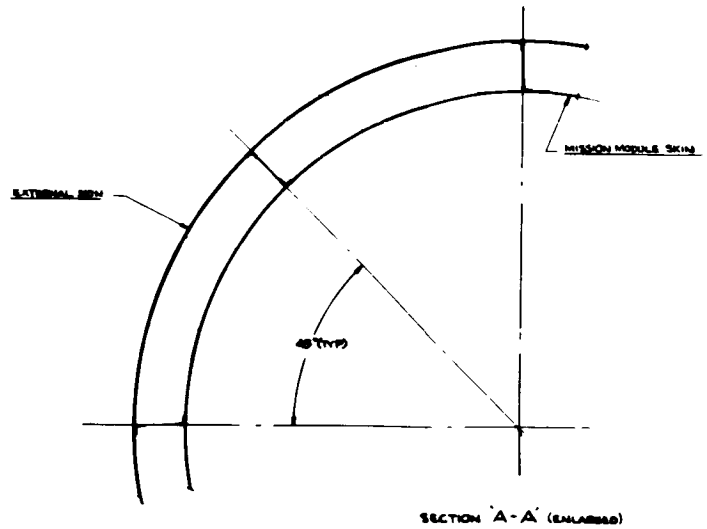
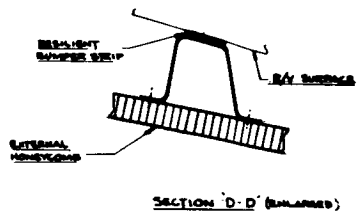
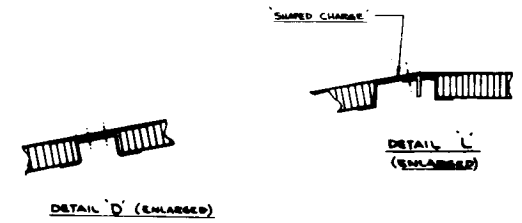


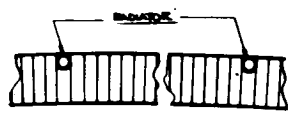
Figure XVI-2-16. Structural arrangement

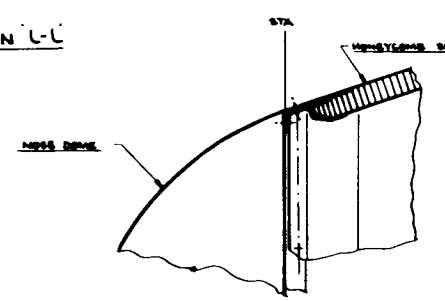

HONEYCOMB
R/V SURFACE
SECTION 'C - C'

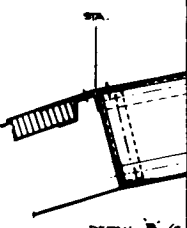

MISSION MODULE SKIN
EXTERNAL SKIN
45° (TYP)
SECTION 'A-A' (ENLARGED)
LONGERON LOCATION
(TYP)

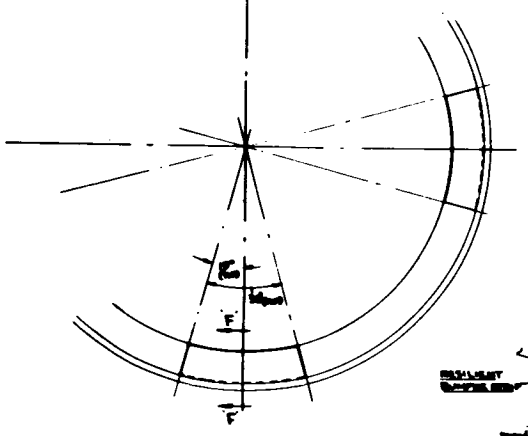

R/V SURFACE
EXTERNAL HONEYCOMB
SECTION 'D-D' (ENLARGED)

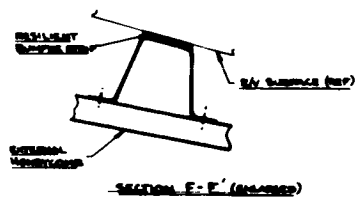

SHAPED CHARGE
DETAIL 'L'
(ENLARGED)
DETAIL 'D' (ENLARGED)

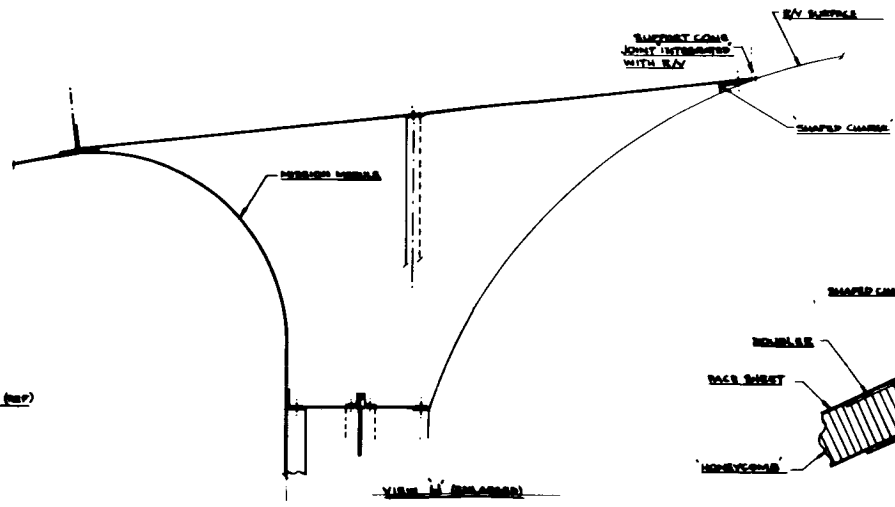

RADIATOR
SECTION 'L-L'

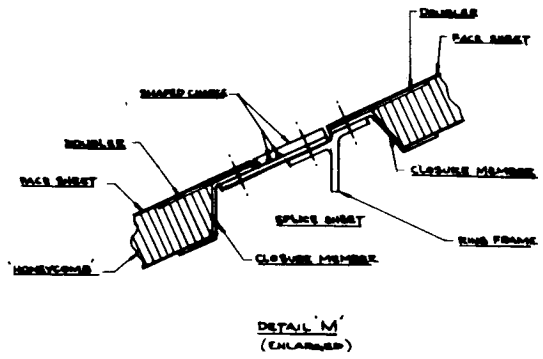

STA.
HONEYCOMB SKIN
NOSE SKIN
DETAIL 'A' (ENLARGED)

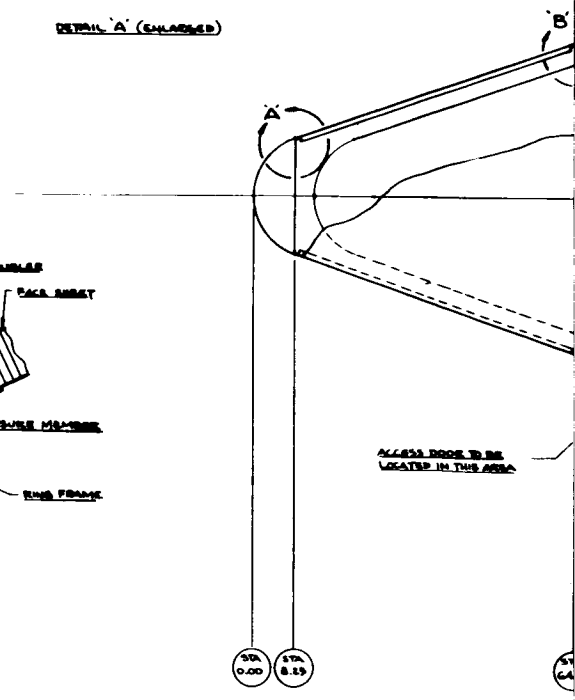

STA.
DETAIL 'B' (E


F
SECTION 'E - E'


R/V SURFACE (REF)
EXTERNAL HONEYCOMB
SECTION 'F-F' (ENLARGED)


R/V SURFACE
SHAPED CHARGE
MISSION MODULE
VIEW 'H' (ENLARGED)


DOUBLER
FACE SHEET
SHAPED CHARGE
CLOSURE MEMBER
SPLICE SHEET
RING FRAME
HONEYCOMB
DETAIL 'M'
(ENLARGED)


B
A
ACCESS DOOR TO BE
LOCATED IN THIS AREA
STA. 0.00
STA. 8.25

FOLD OUT #1

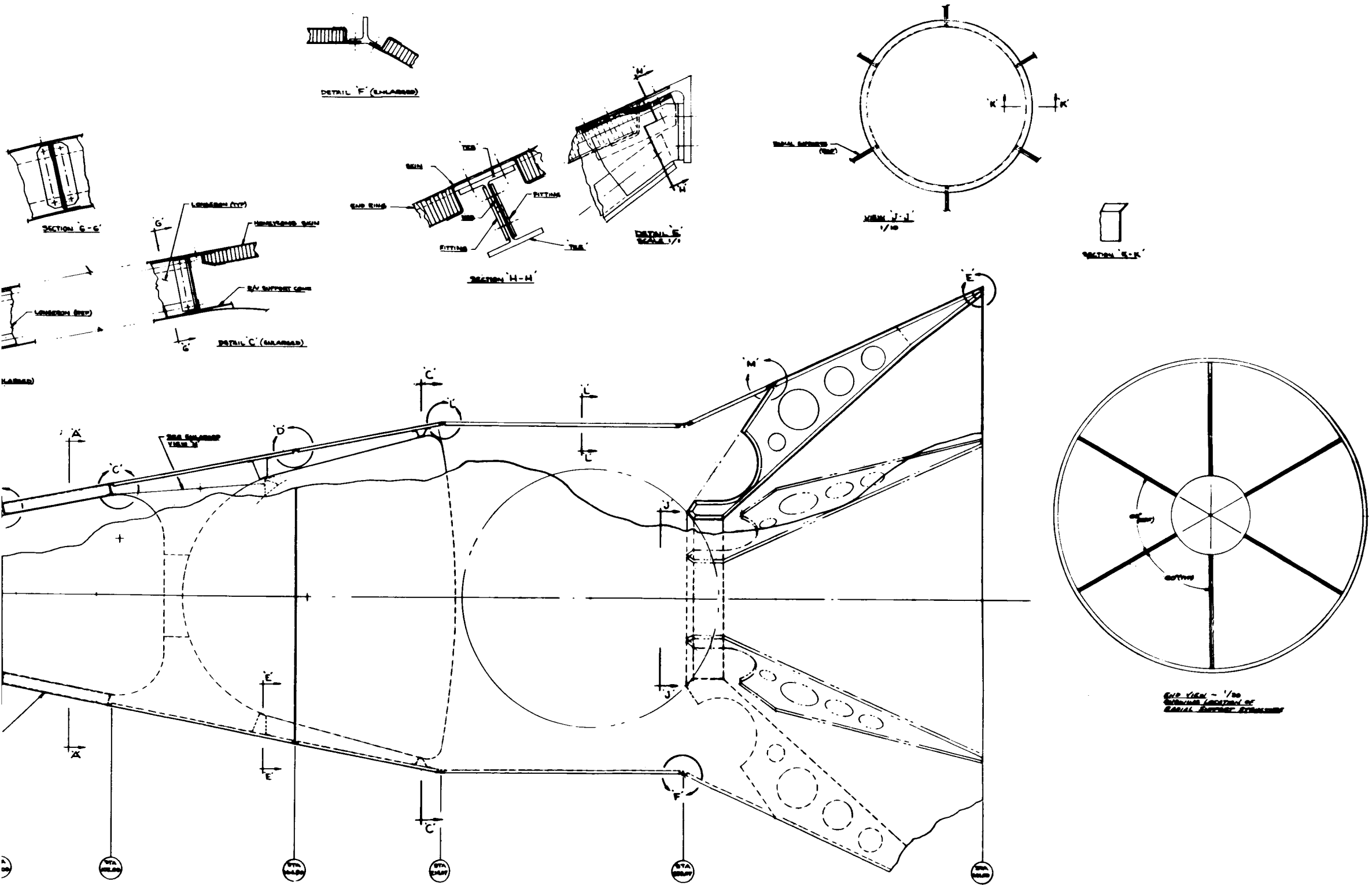


Figure XVI-2-17. Structural arrangement drawing

FOLD OUT #2

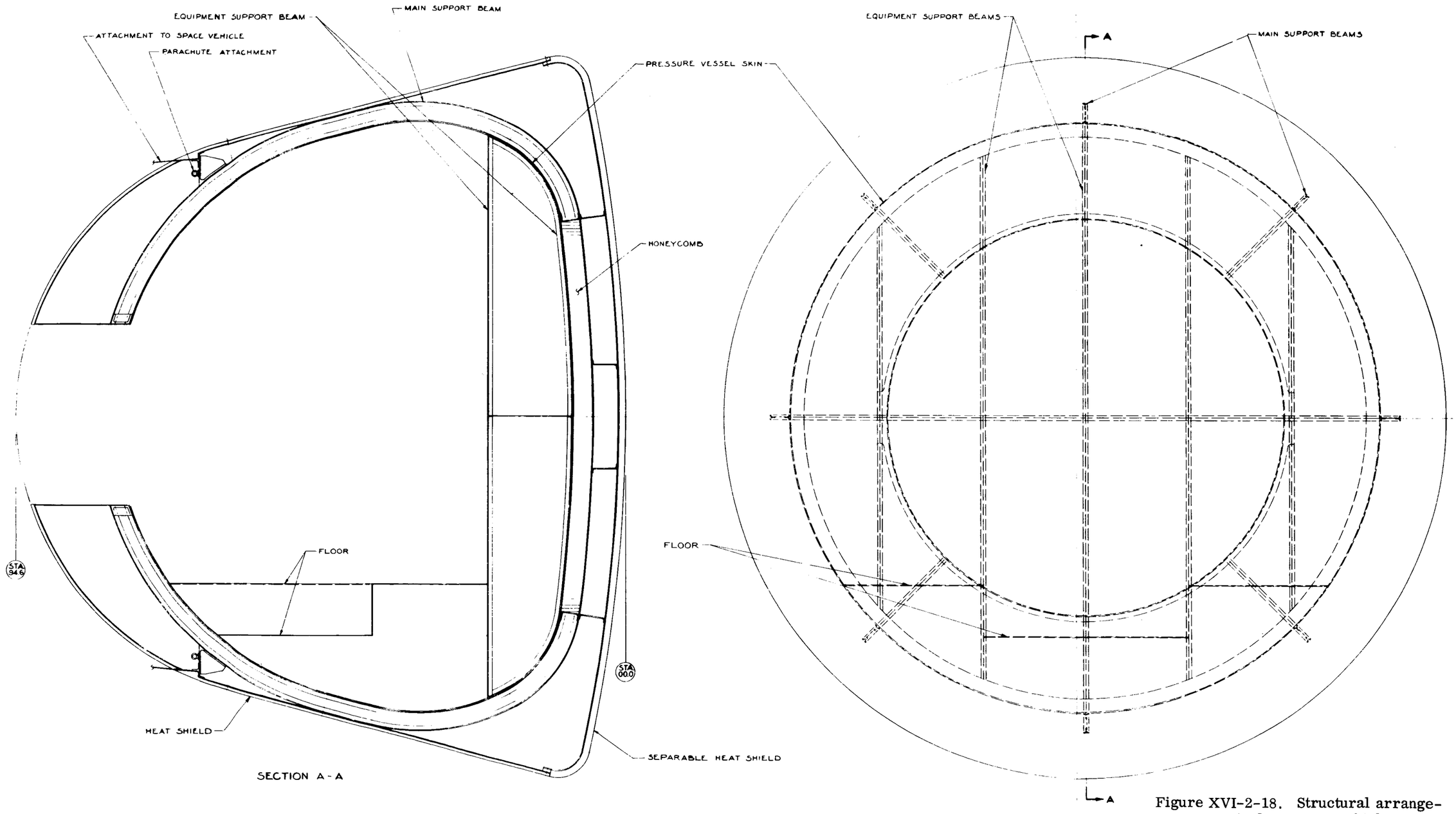


Figure XVI-2-18. Structural arrangement of recovery vehicle

with magnesium. The mission module is constructed of fusion welded 6061 aluminum alloy and either spot welded or bonded to the attaching structure. To minimize leakage, no riveted or bolted joints are used in either the mission module or R/V pressure shells. Hatches are constructed of rigid stiffened plates and employ multiple pressurized seals to further reduce leakage.

The R/V structure consists of the welded 6061 pressure shell and externally bonded or spot welded 2024 aluminum rings and frames. The phenolic nylon or G. E. Series materials heat shield is bonded to an aluminum liner which is in turn attached to the inner R/V structure.

For the large-diameter sections requiring separation on the APOLLO vehicle, the shaped charge separation system is proposed. This system uses a ring of preformed RDX explosive of predetermined cross sectional area and shape to completely sever the shell structure circumferentially.

Extremely small quantities of explosive may be used, with proper development of shape, and the inherent stability of the explosive virtually eliminates danger during and after installation with proper interlocks, shorting devices, and safe-arm initiators. This type of separation device is currently used on the Minuteman program and has been employed extensively in warhead applications. Multiple charges would be employed to assure reliability of separation, and an aluminum back-up ring containing the charge would be employed to minimize internal blast and fragmentation effects. The back-up ring may not be required if the proper charge is developed for the desired blast propagation. Advantages of the shaped charge are much lighter weights than the Marman Clamp system and higher reliability over multiple explosive bolts or mechanical linkage systems. In addition, the shaped charge does not interfere with the shell structural continuity as would local bolt fittings and load redistribution members.

The solar collector is shown on Figure XVI-2-19. The collector assembly consists of three basic components, the solar cell panel, a reflector, and a collimator. This primary assembly is deployed and supported by a telescoping central support column stabilized by three or four support cables, as shown on Figure XVI-2-20. The solar cell panel and collimator are of rigid aluminum construction and the large parabolic reflector is a collapsible aluminum-coated mylar structure. The reflector is extended,

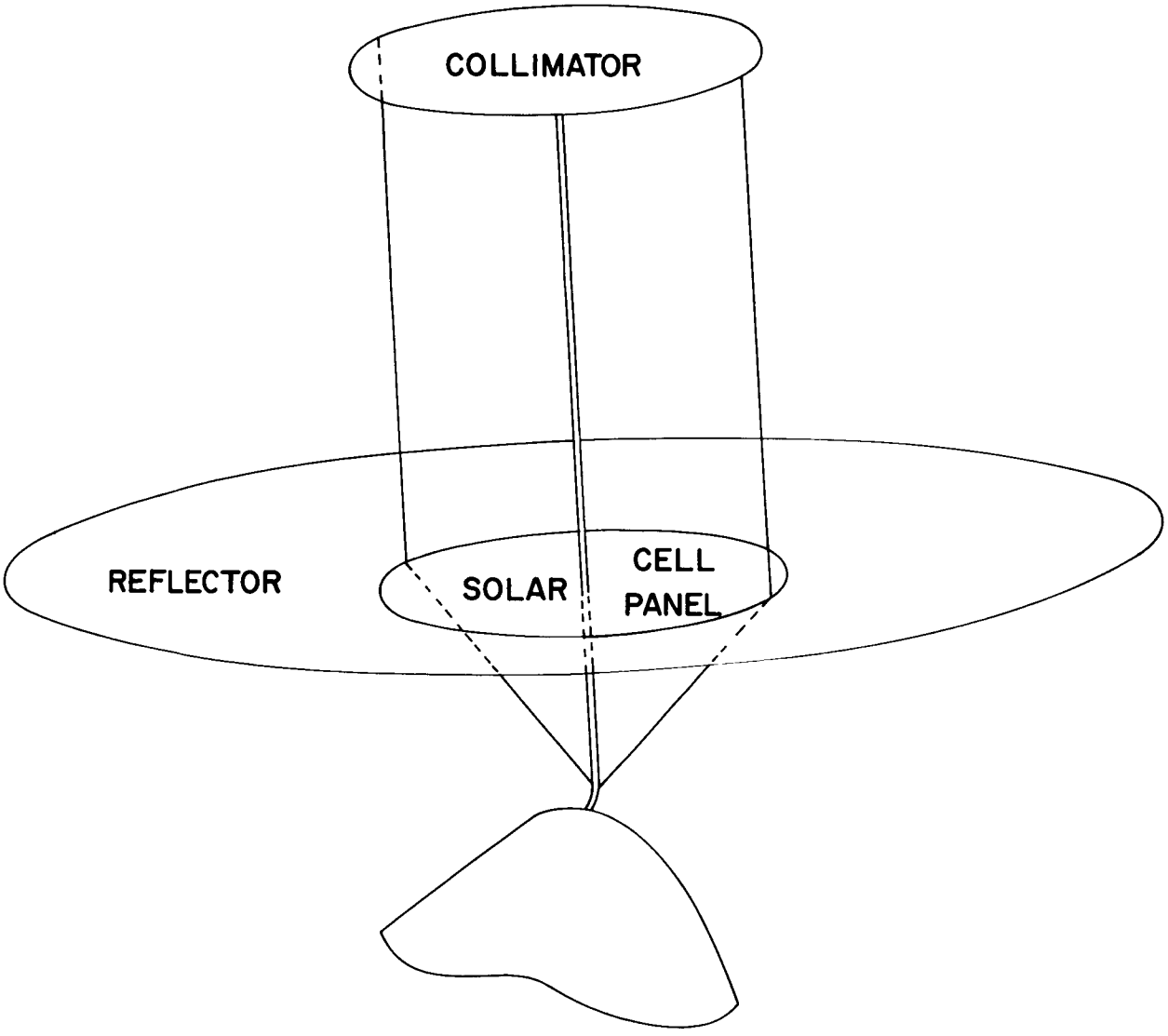


Figure XVI-2-19. Solar collector

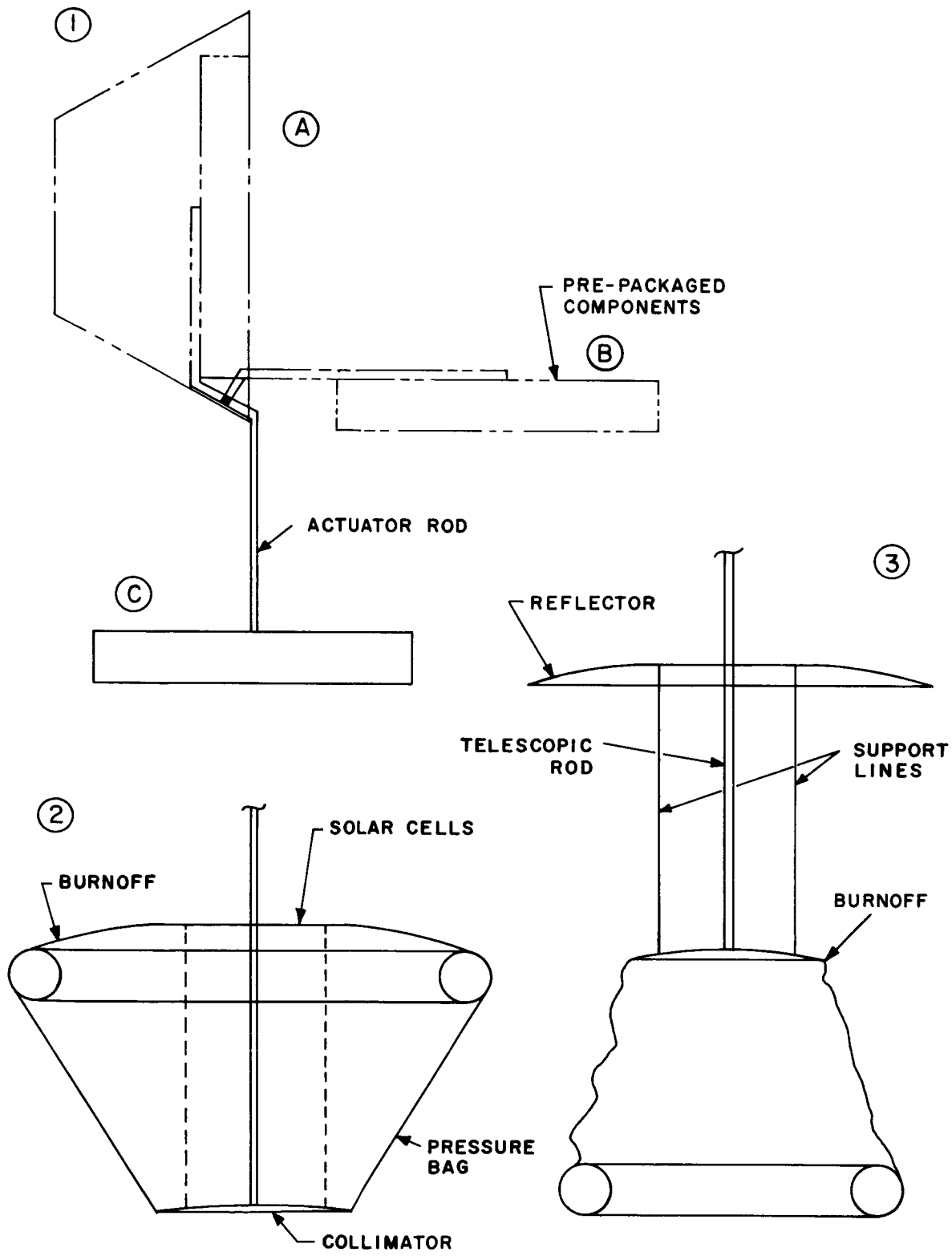


Figure XVI-2-20. Solar collector sequence

after deployment of the assembly, by inflating a mylar bag. The reflector surface is rigidized by a heat-hardenable coating or by a foam-in-place back-up. After rigidizing, the inflation bag is burned off, leaving the operational collector configuration.

Design Criteria

CRITICAL DESIGN CONDITIONS

The conditions listed below and in Table XVI-2-2 have been established for preliminary sizing of the vehicle structure and weights analyses have been made from the resulting limit load distributions summarized on Figure XVI-2-21. A design ultimate factor of safety of 1.5, and design yield factor of 1.15 has been used for all design analysis.

(a) Launch - Max q condition (M = 3 at 30,000 feet, t = 72 sec) plus 100 fps side gust. No relieving lateral inertia is assumed acting for this condition due to booster dynamic response to the initial gust loading. Condition results in maximum bending and shear at the booster - S/V interface.

(b) Abort - Max. rocket thrust equivalent to 20 g plus shock wave from exploded booster. Results in critical collapse pressures on S/V shell, and max. compression in forward section of shell.

(c) Re-entry - Max. combination of normal and axial forces critical for structural design of R/V shell. Maximum deceleration is limited by human tolerances.

(d) Impact - Critical for forward R/V structure and impact attenuation system.

ENVIRONMENTAL CRITERIA

(a) Booster - Shock and vibrations from the booster will establish requirements for the design, mounting, and qualification of all components and structure. Predicted levels are below the values currently applicable in ballistic missile weapon systems design.

(b) Space Environment - Effects of high vacuum, radiation, meteoroids, and temperature extremes will be considered in all phases of design and qualification of structure, materials, and components.

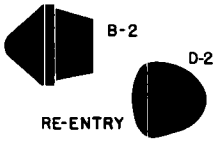
VEHICLE	COND	DESCRIPTION	N AFT	N VERT.	WT KIPS	MACH NO.	ALT 1000	CRITICAL STRUCTURE
LAUNCH	I	BOOST 1) MAX "q" 100 FPS GUST	2.0	1.0	16.6	1.3	30	SHELL & TRANSITION STRUCTURE (SHEAR & BENDING) AFT OF STA 160
		2) MAX AXIAL ACCEL.	5.1	1.0	16.6	9.5	445	PROPULSION PACKAGE SUPPORT STRUCTURE
ABORT IN BOOST PHASE	II	MAX "q" 100 FPS GUST	12.0	5.5	9.0	1.3	30	SHELL SHEAR & BENDING STRUCT FWD STA 0 TO 150
	III	ABORT OFF LAUNCHING PAD MAX AXIAL ACCEL.	20.0	—	9.0	0	S.L	1) MAX SHELL COMPRESSION STA 60 TO 280 2) CRITICAL FOR R/V & MISSION MODULE SUPPORT KEELS & TIE DOWN STRUCTURE TO SHELL
	IV	RE-ENTRY MAX "q" LIFT α = 40°	11.8	2.8	6.0	26.5	150	1) FLAP & FLAP SUPPORT STRUCT. 2) MAX LOCAL PRESSURE ON SHELL
	V	RE-ENTRY MAX "q" DRAG α = 0°	12.0	0	6.0	26.5	150	MAX COLLAPSE LOAD ON FWD SHELL - CRITICAL FOR FWD SHELL FRAMES

TABLE XVI-2-2. SUMMARY OF DESIGN CONDITIONS

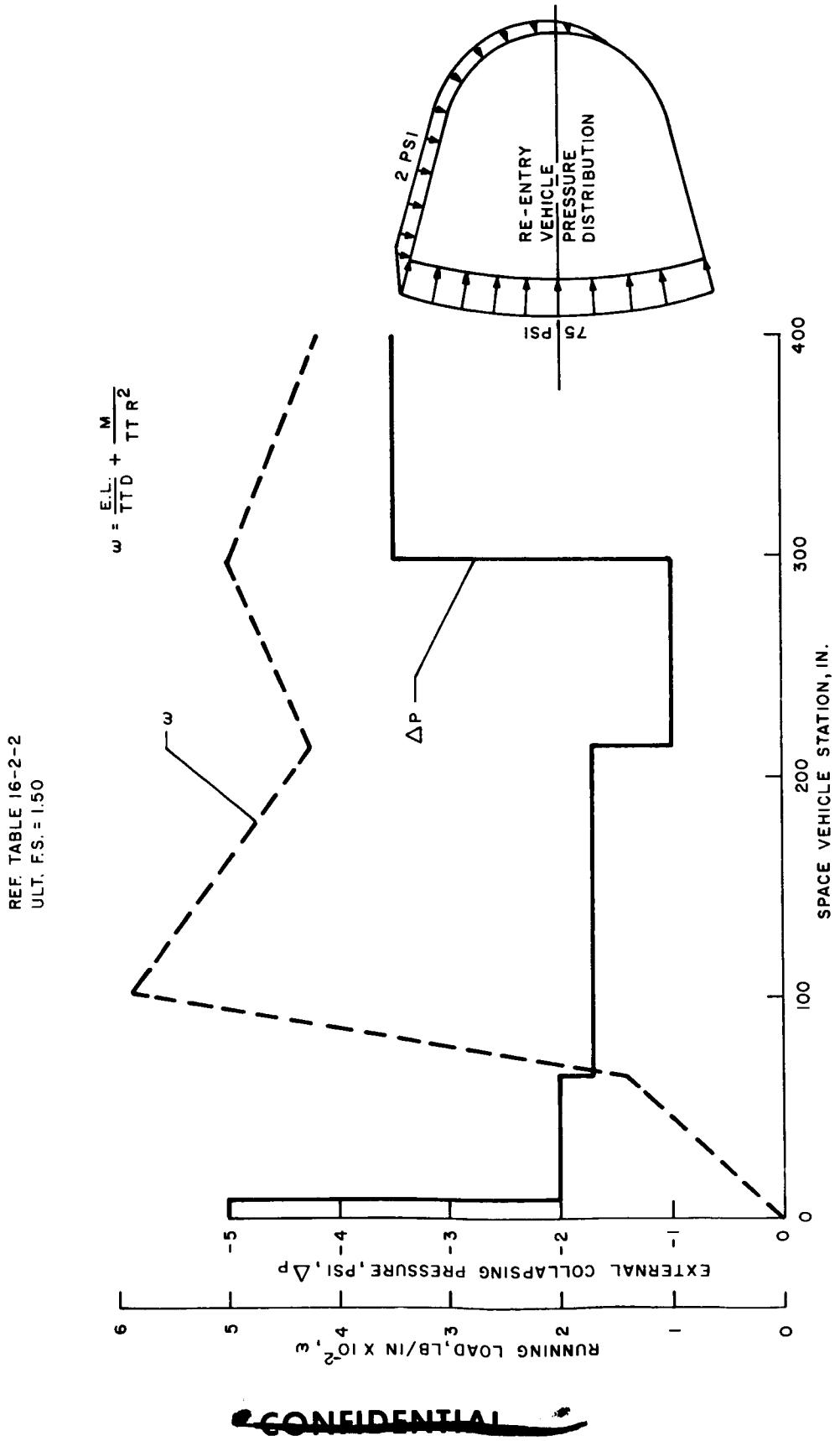


Figure XVI-2-21. Limit collapsing pressure and running load

Present testing of structural materials for the Advent Program indicates that, for the APOLLO exposure time of 15 days, no significant effects of vacuum or radiation exposure is expected for either organic or inorganic materials.

METEOROID PROTECTION

An estimate of the meteoroid population of space is given in Figures XVI-2-22 and XVI-2-23. Protection or suitable repair procedures must be provided to assure survival of space craft, in the event of collision with such particles. The effect of such collision on the APOLLO vehicle has been analyzed by two methods. The first method considered has been developed by M. Kornhauser of G. E. and assumes a meteoroid population equivalent to the Hughes maximum mass data of Figure XVI-2-22. The second, by D. Beard, ARS Journal, Vol. 31, No. 1 Jan. 1961, is based on an average meteoroid population. Results of these analyses illustrate the variations in damage predictions and the necessity of careful consideration of meteoroid damage in space vehicle design. Figure XVI-2-24 shows a comparison of both methods as a function of skin thickness.

The results shown on Figure XVI-2-25 reflect the "bumper" effect of the vehicle outer shell structure in protecting the internal components by shattering small meteoroids penetrating the shell. The Kornhauser method indicates the overall vehicle shell would be penetrated 30 times by meteoroids, but the internal equipment would receive less than one penetration during the mission. The D-2 R/V pressure capsule, protected by multiple bumpers (outer shell and R/V structure) would receive one penetration in 100 missions, and the R-3 vehicle cabin one penetration in 20 missions (due to the small local cabin area exposed).

The more optimistic (but not necessarily more valid) method of Beard gave 10 penetrations of the vehicle shell, one penetration of the critical areas in 10 missions, and only one penetration in 1000 missions for the D-2 pressure capsule. The R-3 cabin would be penetrated once in 250 missions.

These results have been based on a rational assessment of the effect of bumpers in alleviating meteoretic damage based on extensive tests conducted by G. E. using a hypervelocity gun firing 1/4-inch-diameter steel and aluminum pellets at velocities

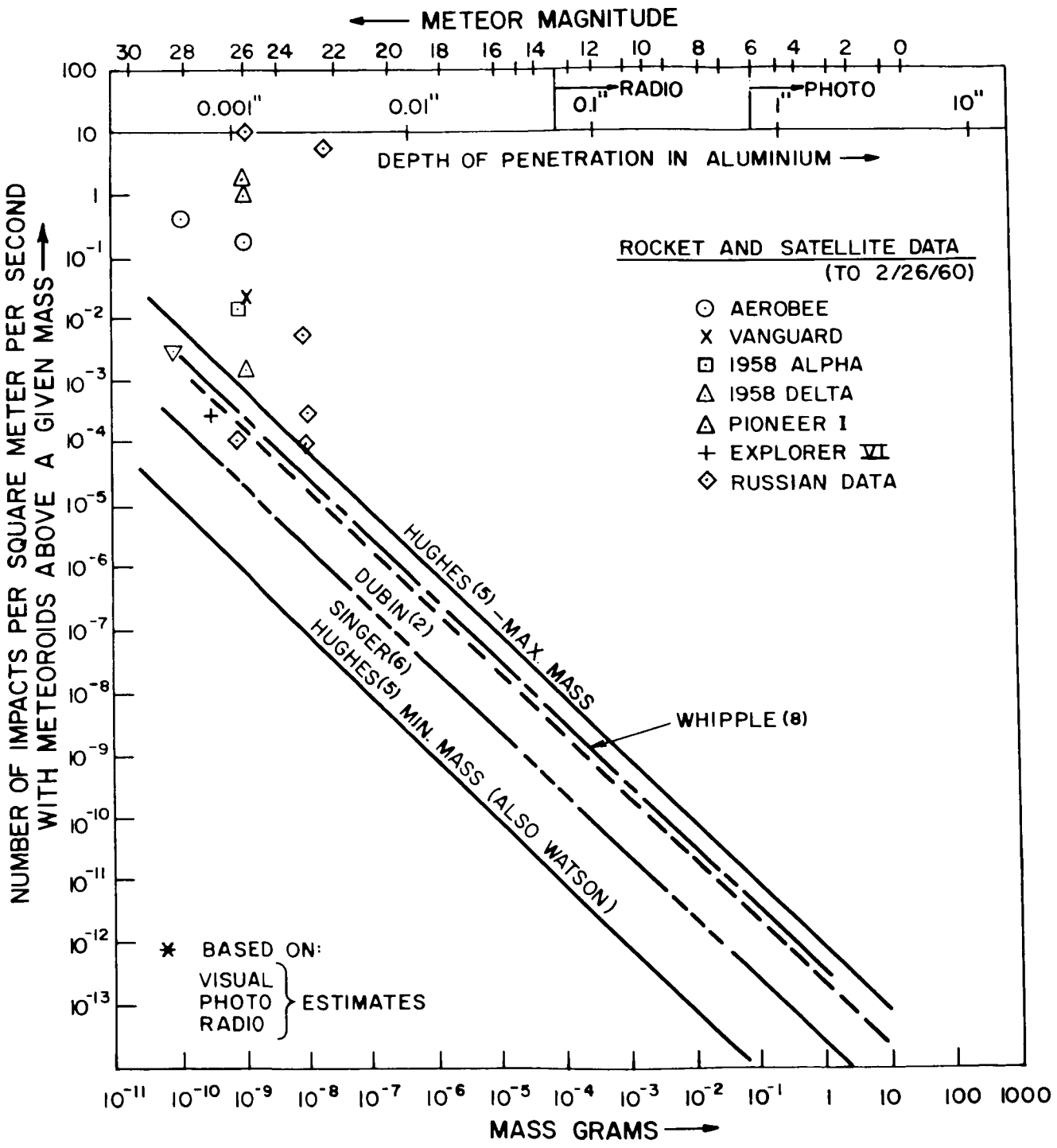


Figure XVI-2-22. Expected impacts per second per square meter of surface area with meteoroids above a given mass versus mass of meteoroid

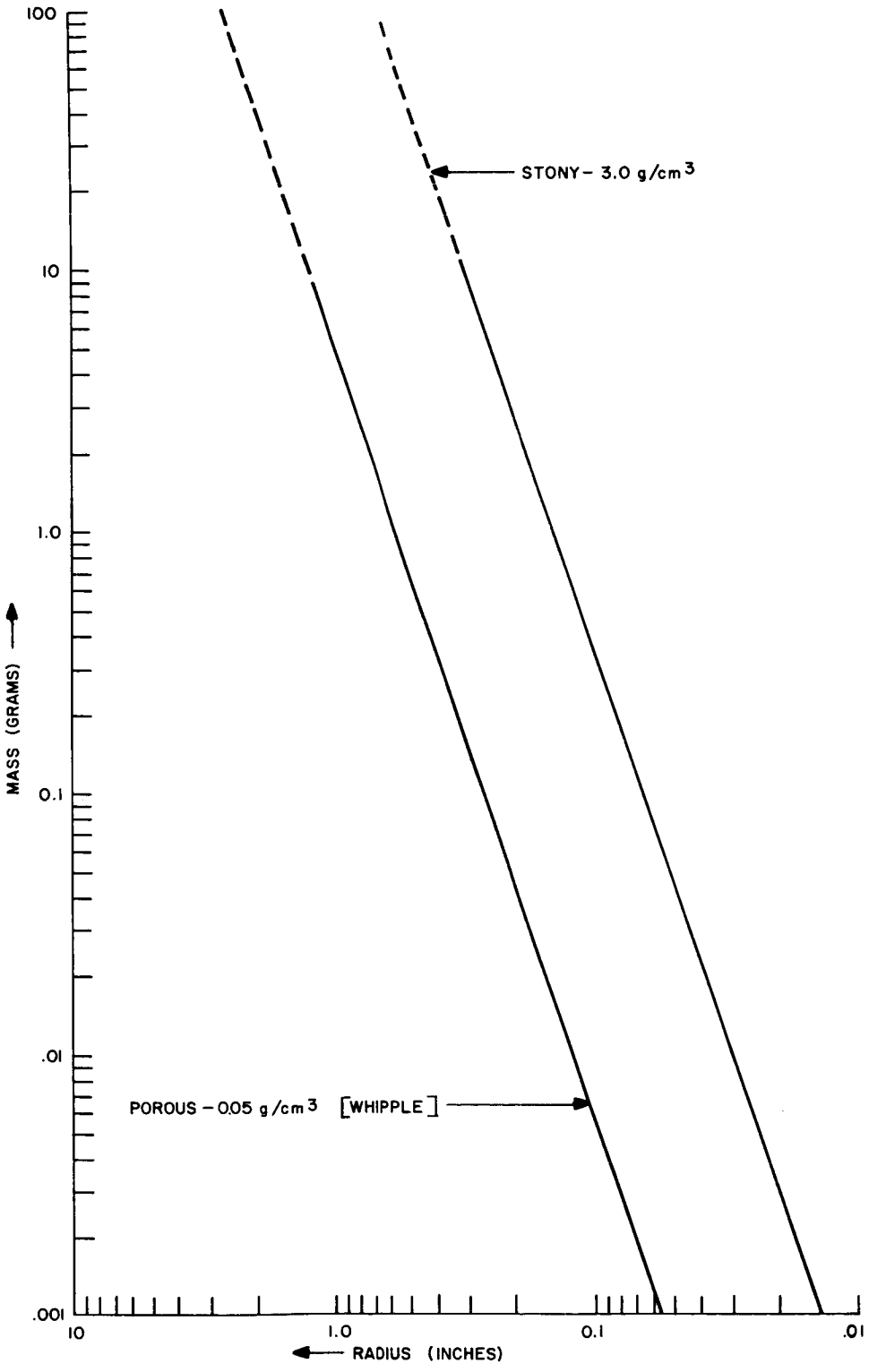


Figure XVI-2-23. Size versus mass of meteoroids

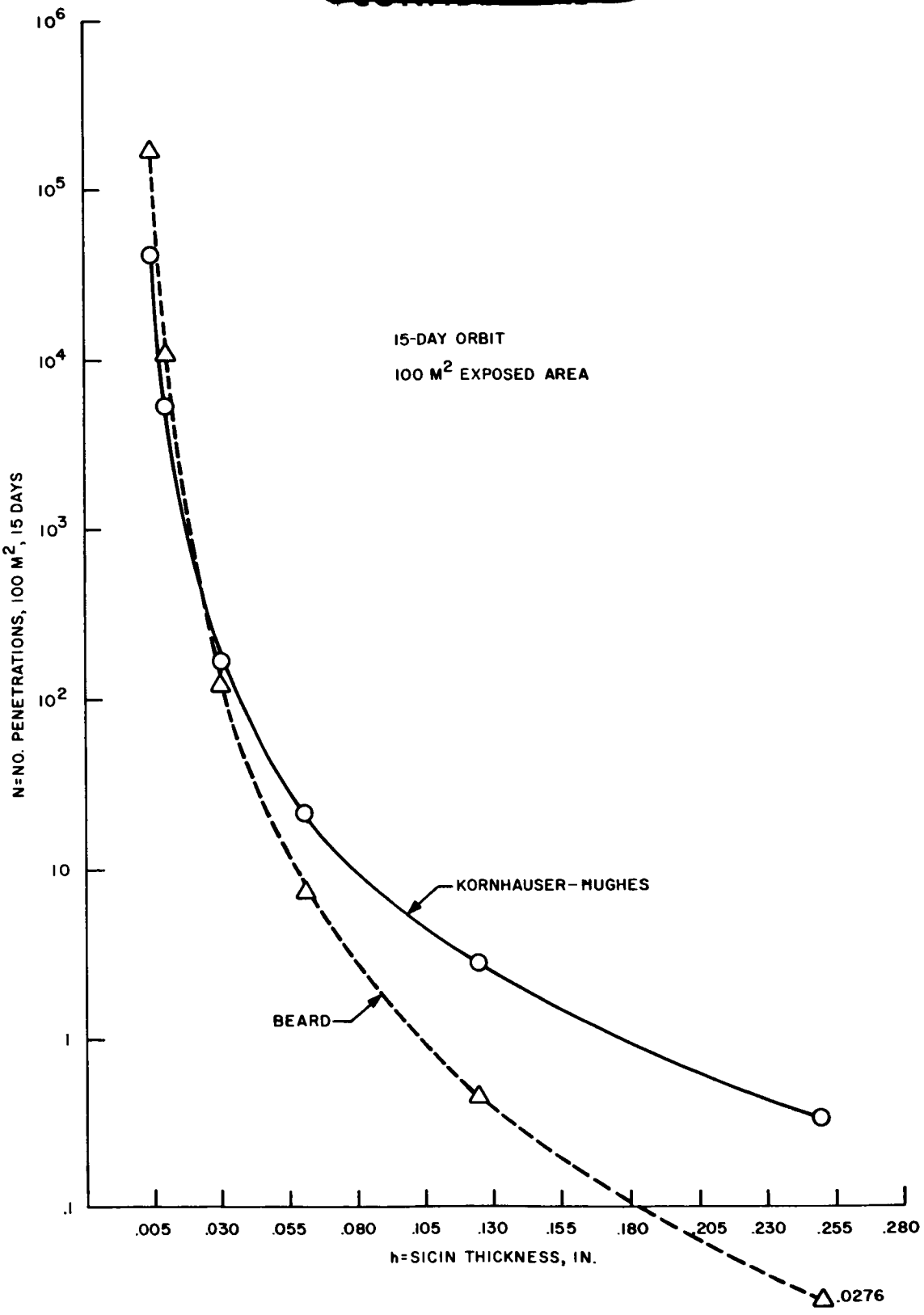


Figure XVI-2-24. Meteoroid penetration analysis (single wall)

METHOD / VEHICLE AREA		KORNHAUSER-HUGHES MAX MASS DATA	BEARD-AVERAGE METEOROID POPULATION
OVER ALL VEHICLE		30	10
CRITICAL AREA (BUMPER PROTECTED)		.8	1/10
R/V CAPSULE (MULTIPLE BUMPER PROTECTED)	D_2	1/100	1/2500
	R_3	1/20	1/1000

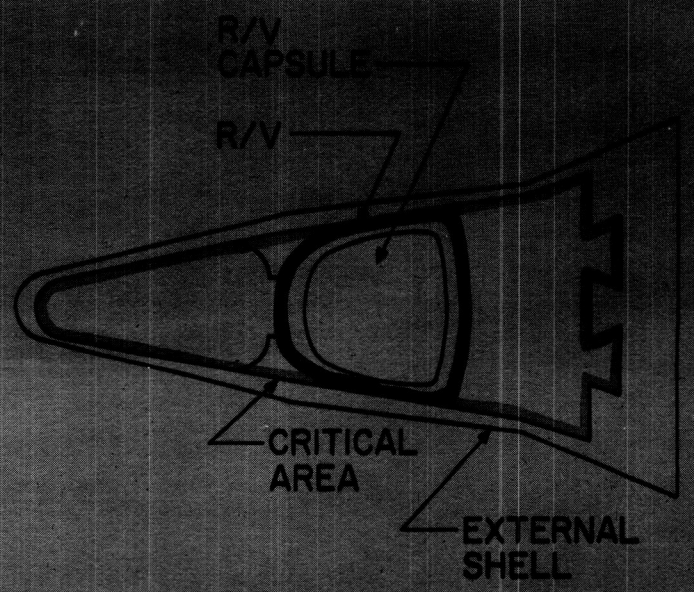


Figure XVI-2-25. Meteoroid damage — predicted number of penetrations for 15-day mission

of 17,000 fps. Note that the multiple bumper effect provided by the D arrangement provides added resistance to penetration over the more exposed single-bumper B-3 vehicle. Both vehicles, however, have sandwich construction outer surface structure, which should provide a maximum degree of protection for a given weight due to the double skin construction.

Minimum penetration hole size is roughly twice the skin thickness, and as probability of penetration decreases due to either thicker skins, or added protection, the hole size will increase.

Estimates of penetration hole size for the D pressure-capsule is approximately 1/2 inch or larger, and serves to emphasize the necessity for access to the capsule inner surface for repair if required.

Multiple tanks for mid-course and lunar orbit, disorbit propulsion, and redundant rocket motors and critical subsystems increase mission reliability. Note that no bumpers were considered acting on the vehicle open aft end in determining penetration of critical areas. Local bumpers (a good use for beryllium) could further reduce meteoroid damage.

References:

Kornhauser, M., "Current Estimates of the Effects of Meteoroids on the Skin of a Satellite Vehicle," General Electric, MSVD, Advanced Space Vehicle Engineering Memo #3, February 26, 1960.

Beard, D. B., "Meteoritic Impact", ARS Journal, Vol. 31, No. 1, January, 1961.

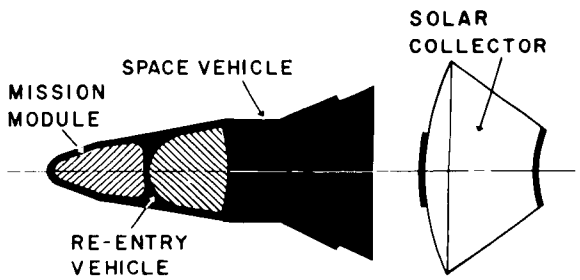
SUMMARY OF STRUCTURAL CHARACTERISTICS

Table XVI-2-3 summarizes the materials, type of construction, and development status for the proposed APOLLO structure.

Heat shield materials are either phenolic nylon currently in use on the Mark 3 ballistic R/V program, or the G.E. Series castable materials under development for the Mark 6 R/V Program. These materials have been flight-tested on the Mark 3 and RVX-2

TABLE XVI-2-3. SUMMARY OF STRUCTURAL CHARACTERISTICS

	ITEM	MATERIAL	CONSTRUCTION	MAX. LOAD	MAX. TEMP.	DEVELOPMENT STATUS
RE-ENTRY VEHICLE	HEAT SHIELD	GE SERIES 100 OR 500	MOLD-CAST	RE-ENTRY	RE-ENTRY	RVX FLIGHT TESTED. MARK VI DEVELOPMENT
		PHENOLIC NYLON	BONDED TAPE WOUND			IN HOUSE USAGE ON NERV, RVX & MARK III
	OUTER-SHELL	ALUMINUM MAG-THOR	CONVENTIONAL SHEET METAL	RE-ENTRY (PRESSURE)	RE-ENTRY	STATE OF THE ART
	INTERNAL STRUCTURE	ALUMINUM	WELDED SHELL BONDED SHEET METAL FRAMES	ABORT & LANDING IMPACT	ENVIRON-MENT CONTROL 70° F	STATE OF THE ART MATL'S. DEVELOPMENT OF PRESSURE SEALS, & SELF SEALANT MAT'L. & REPAIR TECHNIQUES FOR PUNCTURES
MISSION MODULE	SHELL & SUPPORT STRUCTURE	ALUMINUM	SAME AS ABOVE	POWERED FLIGHT & ABORT	ENVIRON-MENT CONTROL 70° F	SAME AS ABOVE
SPACE VEHICLE	OUTER-SHELL	ALUMINUM HONEYCOMB	BONDED ALUMINUM CORE & FACES	POWERED FLIGHT & ABORT	2ND STAGE BOOST	GENERAL USAGE AT PRESENT DEVELOPMENT OF LARGE BONDED ASSEMBLIES
	INTERNAL SUPPORT STRUCTURE	ALUMINUM OR MAGNESIUM	CONVENTIONAL SHEET METAL	POWERED FLIGHT & ABORT	2ND STAGE BOOST	STATE OF THE ART POSSIBLE BERYLLIUM USAGE
SOLAR COLLECTOR	PARABOLIC REFLECTOR WITH SOLAR CELLS	EPOXY HARDENED SHELL WITH ALUMINIZED SURFACE	SPIN MOLD FORMED WITH INFLATABLE MYLAR BAG	POWERED FLIGHT	LUNAR SPACE FLIGHT	1.) SPACE INFLATABLE STRUCTURE & REFLECTOR DEVELOPMENT 2.) RELATED IN HOUSE PROGRAMS: STEPS, SOLAR SIMULATOR & AIR FORCE - MSVD R&D ON INFLATABLE SOLAR REFLECTORS



programs and represent present design capability. Subsequent development of the G.E. Series materials may result in increased weight savings in addition to the lower production costs.

Space vehicle and R/V internal structural materials are conventional aluminum and magnesium alloys and the outer shell of aluminum honeycomb, is in widespread use.

WEIGHT ANALYSIS

Overall vehicle weight estimates and a detailed weight breakdown are shown in this section.

Table XVI-2-4 shows the subsystem weights for both present and predicted 1963 design capability. The 1963 weight predictions are based on state-of-the-art improvements such as limited use of beryllium for sub-structure, heat shield materials improvements, propulsion system optimization, and miniaturization in all subsystem areas.

Table XVI-2-5 lists the distribution of present subsystem weights between the re-entry vehicle (R/V), space vehicle (S/V), and abort configuration. Note that a significant portion of the total weight remains with the booster for on-the-pad abort. Preliminary center of gravity locations and moments of inertia are shown in Table XVI-2-6 for the present vehicle design.

Table XVI-2-7 is a detailed weight breakdown of the subsystem components summarized in the previous tables.

TABLE XVI-2-4. WEIGHT SUMMARY, "D" CONFIGURATION

Subsystem	Present Design Capability Weight	Predicted '63 Design Capability Weight
Propulsion	7620	7200
Life Support	1909	1700
Nav. & Control	1065	850
Power Supply	782	630
Search & Recovery	625	550
Electrical Sys.	285	250
Instr. & Comm.	159	150
Structure	4031	3650
Gross Weight	16476	14980

TABLE XVI-2-5. WEIGHT BREAKDOWN, "D" CONFIGURATION

Subsystem	R/V	S/V (Total)	S/V (Abort)
Structure	1953	2078	968
Instr. & Comm.	107	52	36
Power Supply	357	425	-
Search & Recovery	625	-	-
Propulsion Sys.	-	7620	600
Nav. & Control	340	725	725
Elect Sys.	75	210	160
Life Support	1358	551	551
Total	4815	11,661	3040

Gross Wt. R/V + S/V = 16,476

Abort Wt. R/V + S/V = 7855

TABLE XVI-2-6. PRELIMINARY WEIGHT & BALANCE STATUS
(D-2 CONFIGURATION)

Parameter	R/V (Only)	Total Veh. (S/V) (Power Flight)	Total Veh. (S/V) Less Abort Rock. & Solar Reflect. & Ant. Extended
Weight (Lbs.)	4815.0	16476.0	15876.0
Center of Gravity (In.)	-	-	-
Roll (Long) (From Sta. 0)	① 34.7	③ 235.4	③ 232.2
Yaw (Vert.) (From ℄)	-	-	5.5
Pitch (Lat.) (From ℄)	-	-	-
Mom. of Inertia (Slug-Ft.^{2})	-	-	-
Roll (About ℄)	860.0	4316.5	14225.5
Yaw (About C. G.)	②1051.0	④20695.0	④32332.0
Pitch (About C. G.)	②1051.0	④20695.0	④30068.0
Prod. of Inertia (Slug-Ft.^{2})	-	-	-
Roll - Yaw	Negligible	Negligible	11705.0
Roll - Pitch	Negligible	Negligible	Negligible
Pitch - Yaw	Negligible	Negligible	Negligible

① From R/V Sta. 0

② About R/V C. G.

③ From S/V Sta. 0

④ About S/V C. G.

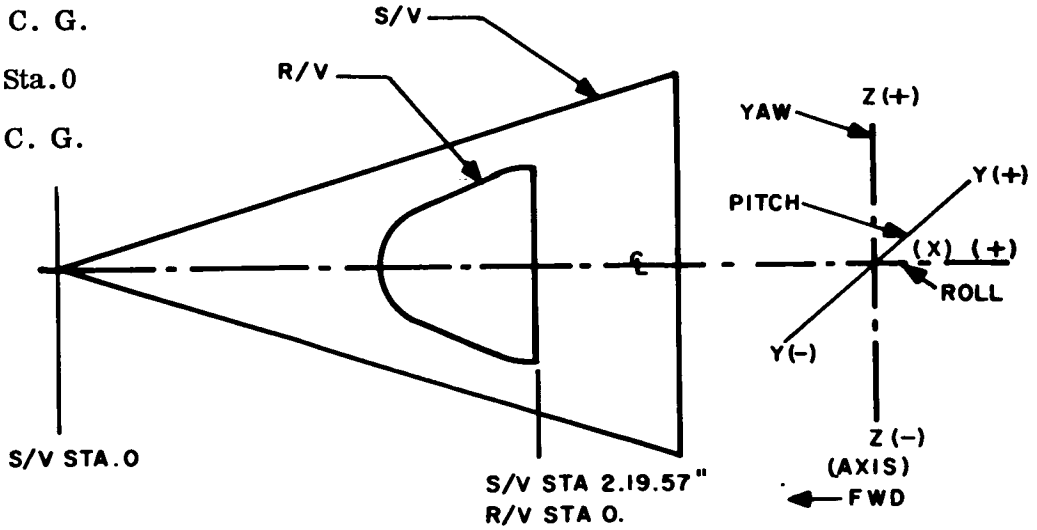


TABLE XVI-2-7. DETAILED WEIGHT BREAKDOWN
D-2 CONFIGURATION

STRUCTURE				4031.0
R/V			1953.0	
Heat Protection		1155.0		
Shield Supt's.		68.0		
Separation Ring & Charge		40.0		
Afterbody Skin		96.0		
Pressure Vessel Skin		135.0		
Supt. Beams		45.0		
Afterbody Bulkhd.		64.0		
Afterbody Ftg's.		45.0		
Seal Hatch		24.0		
Flaps		250.0		
Floor		18.0		
Misc'l. & Etc.		13.0		
S/V			2078.0	
External Shell (Honeycomb)	=	780.0		
Heat Protection Coating		190.0		
Mission Module Skin		136.0		
Frames & Long		238.0		
Separations & Misc'l.		359.0		
Propulsion Supts.	=	375.0		
SUBSYSTEMS				
INSTRUMENT & COMMUNICATIONS				
R/V SECTION			107.0	159.0
2 KMC-2 Watt Pr. XMTR. (3)		15.0		
400 MC - 3 Watt XMTR. Rec.		5.0		
Data Recorder		25.0		
2 KMC Rec. Decoder (2)		10.0		
PMC Multiplexer		38.0		
400 MC 3 Watt XMTER.		6.0		
Antenna (2)		8.0		
SPACE (MISSION) VEHICLE			52.0	
2 KMC 16 Watt PWR. Amp.		15.0		
Multiplexer		4.0		
T. V. Camera		17.0		
Antenna		16.0		
POWER SUPPLY SYSTEM			357.0	782.0

TABLE XVI-2-7. DETAILED WEIGHT BREAKDOWN
D-2 CONFIGURATION (Continued)

R/V SECTION			
Batteries (Emer.)	103.0		
Fuel Cells	190.0		
Regulators (16)	64.0		
SPACE (MISSION) VEHICLE		425.0	
Solar Cells & Reflector	425.0		
PROPULSION SYSTEM			
SPACE VEHICLE			7620.0
Abort System	600.0		
Thrust Chamber (Tanks Incl'd.)	790.0		
Fuel, Oxid. & Press.	6230.0		
NAVIGATION & CONTROL			1065.0
R/V SECTION		340.0	
R/V Flt. Control Sys.	100.0		
Stable Plat.	75.0		
Platform Elect.	15.0		
Computer	50.0		
Radar Altimeter	50.0		
Navig. Aids	25.0		
Misc'l.	25.0		
SPACE (MISSION) VEH.		725.0	
Attit. Control	60.0		
Manual Navig. Equip.	25.0		
Misc'l.	14.0		
IR Sensor & Elec (2 Req'd)	26.0		
Fuel & Oxidizer	500.0		
Tanks & Misc'l.	100.0		
ELECTRICAL SYS.			285.0
R/V SECTION		75.0	
Elec. Wiring	35.0		
Elec. Equip.	35.0		
Inter-Stage Gnm.	5.0		
SPACE (MISSION) VEH.		210.0	
Elec. Wiring	50.0		
Elec. Misc'l. Equip.	95.0		
Umbilical Conn.	5.0		

TABLE XVI-2-7. DETAILED WEIGHT BREAKDOWN
D-2 CONFIGURATION (Continued)

Inter-Stage Gnm.	10.0		
In-Flt. Conn.	5.0		
Misc'l.	45.0		
LIFE SUPPORT EQUIP.			1909.0
R/V		1358.0	
Crew (3)	540.0		
Acc. Seats & Etc.	90.0		
Survival Equip.	75.0		
Personal Equip.	30.0		
Console (Instr.)	25.0		
Lighting & Trim	5.0		
Misc'l. Furnish	25.0		
Cabin Comm.	5.0		
Q_2 Emerg.	150.0		
Diluent Supply	28.0		
C_{O2} Rem. (Normal & Emerg.)	70.0		
Heat Exchanger (Emerg.)	15.0		
Fire Control	15.0		
Sen. & Control	25.0		
Urine & Feces Stor.	5.0		
Instr. & Contr.	150.0		
Sec. Press Sys.	105.0		
SPACE (MISSION) VEH.		551.0	
Space Radiators	35.0		
Crew Stat. Seat	5.0		
Toilet	15.0		
Galley & Food (H_2O/Incl.)	312.0		
Instr. (Console)	20.0		
Recreation Equip.	1.0		
Lighting & Trim	5.0		
Misc'l. Furn.	15.0		
O_2 Primary	50.0		
Diluent Supply	28.0		
Heat Exchanger	10.0		
Fire Control	15.0		
Maint. Equip. & Misc.	40.0		
RESEARCH & RECOVERY			
R/V SECTION			625.0
Beacon-Elec.	8.0		
Antenna	3.0		

TABLE XVI-2-7. DETAILED WEIGHT BREAKDOWN
D-2 CONFIGURATION (Continued)

Chaff Ejector	8.0
Flare Cluster	9.0
PWR. Supply	15.0
Programmer	6.0
Chaff	2.0
Light	2.0
So-Far Bomb	4.0
Dye Marker	4.0
Main Chute	130.0
Brake Chute	30.0
Chute-Equip.	40.0
Flotation Sys.	150.0
Shock Absorber	204.0
Hand Oper. Dev.	10.0

3. GLIDE VEHICLE DESIGN

SUMMARY

The glide re-entry vehicle constitutes the other logical approach for APOLLO. The factors controlling design for a glide re-entry vehicle include:

Corridor width

Cross-range maneuverability

Re-entry heating

Conventional landing

Gross weight

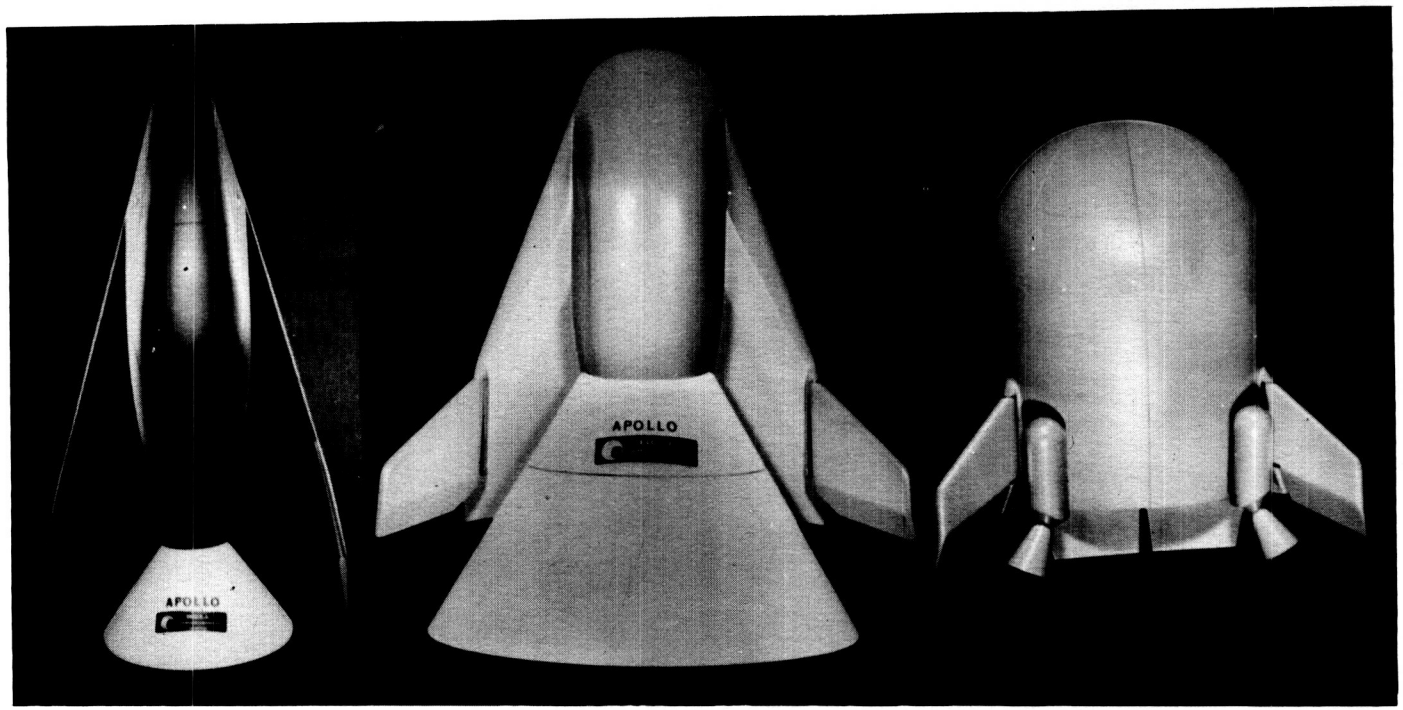
These factors are interdependent and their optimization delineates the flight configuration. The extreme weight limitation and the large lunar propulsion requirement suggests that the re-entry energy absorption be purely aerodynamic. The glider with aerodynamic control seems well suited for these requirements. A group of glide configurations studied are shown in Figure XVI-3-1.

High L/D

Parametric analyses in aerodynamics, propulsion and structures first resulted in the R-1, a high L/D, winged glider. (Figure XVI-3-1-left.) Extrapolation of Dyna-Soar characteristics to re-entry at escape velocity showed that this configuration could give a large re-entry corridor and large cross-range maneuverability. However, high peak heating rate at maximum g re-entry and the high total heat load required excessive heat protection. The weight of the wing and the heat protection put the R-1 configuration out of the competition.

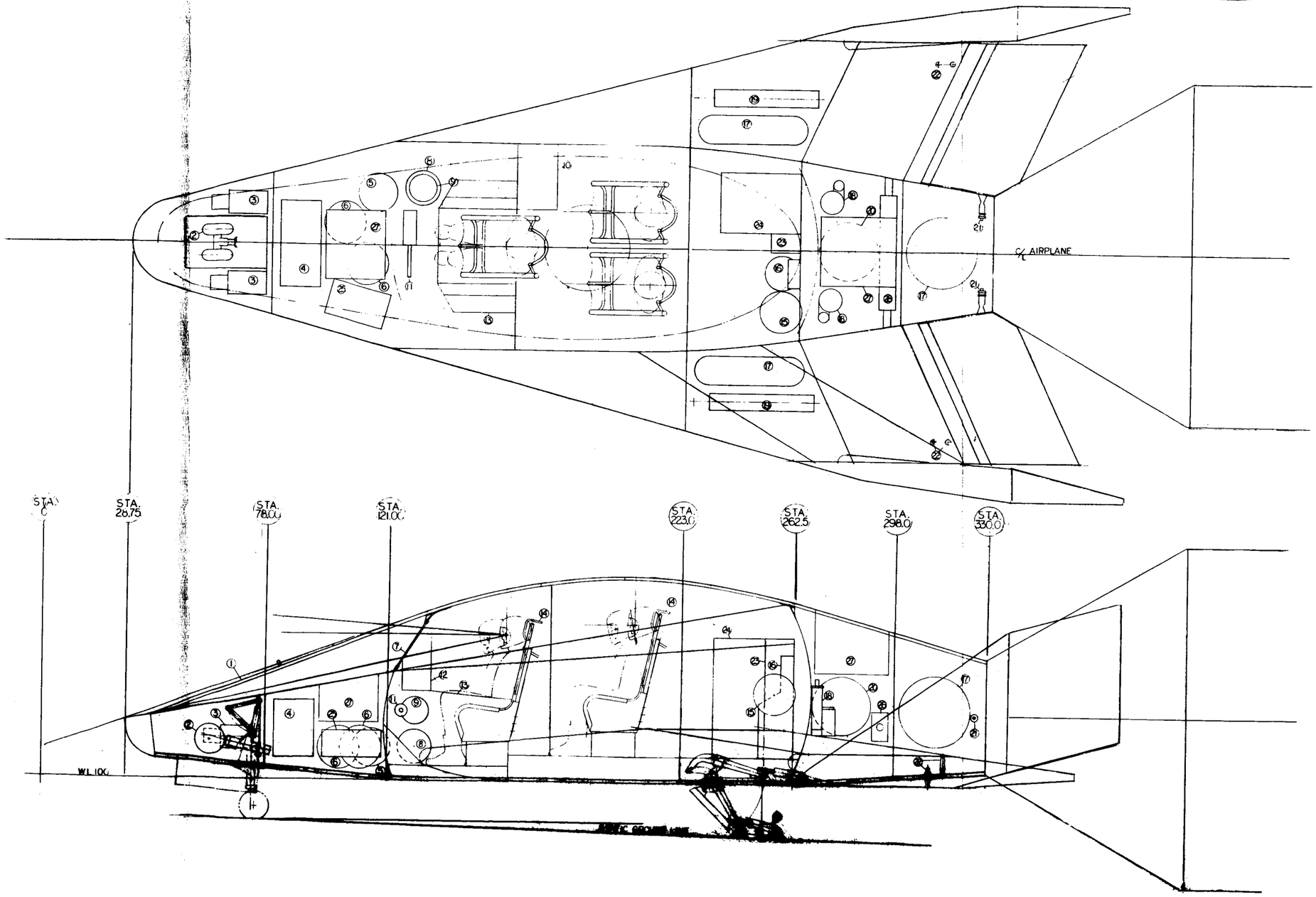
The re-entry vehicle is 29 feet long and 14.5 feet wide. (Figure XVI-3-2.) It has conventional fuselage housing the pilot with two crew members sitting side by side behind him. Wings are swept-back 75 degrees, and vertical stabilizers are attached

CONFIDENTIAL



CONFIDENTIAL

Figure XVI-3-1. Glide configurations. Left to right: R-1 high L/D, R-2 high attitude, R-3 modified lenticular



1. WINDSHIELD HEAT SHIELD (RETRACTABLE)
2. NOSE LANDING GEAR
3. HYDRAULIC PUMP AND MOTOR
4. BATTERIES
5. NITROGEN TANK
6. OXYGEN TANK - AIR CONDITIONER
7. WINDSHIELD
8. HELIUM TANK - AIR CONDITIONER
9. EMERGENCY OXYGEN TANK
10. FOOD AND DRINKING WATER
11. EJECTOR - AIR CONDITIONER
12. INSTRUMENT PANEL
13. CONSOLE
14. CREWMEN SEATS
15. WATER TANK - AIR CONDITIONING
16. FAN, FILTER AND HEAT EXCHANGER - AIR CONDITIONER
17. WATER TANKS - EQUIPMENT AND STRUCTURE COOLING
18. HYDRAULIC RESERVOIR AND ACCUMULATOR
19. MAIN LANDING SKID GEAR
20. HYDROGEN PEROXIDE TANK - REACTION CONTROLS
21. REACTION CONTROL THRUSTERS - YAW
22. REACTION CONTROL THRUSTERS - PITCH AND YAW
23. HEAT EXCHANGER AND FAN - EQUIPMENT COOLING SYSTEM
24. EQUIPMENT COMPARTMENT
    - ELECTRICAL
    - GUIDANCE
    - COMMUNICATIONS
    - TELEMETRY
    - ELECTRONICS
25. DIVERTERS
26. HEAT EXCHANGER- STRUCTURE COOLING
27. PARACHUTES (2)

Figure XVI-3-2. Inboard profile, high L/D glider

at the tip. Elevons provide aerodynamic control during re-entry and landing. The entire fuselage and upper surface of the wing is covered with double-wall construction. The lower surface has an ablative shield over the double-wall construction. A conical transition section provides support for the re-entry vehicle and houses the on-board propulsion system. No mission module is required, making this configuration an integral vehicle.

Reaction controls using bipropellants are contained within the re-entry vehicle. This provides attitude control for the entire lunar mission as well as during re-entry prior to the development of aerodynamic control.

High AttitudeVehicle

To reduce the heat protection weight from that of the high L/D configuration, the R-2 glider was developed. (Figure XVI-3-1, center.) This also is a Dyna-Soar-type vehicle; however, it was designed to fly at high attitudes during super-orbital re-entry. Thus, only the bottom surface was exposed to the extreme heating. The fins were folded in for protection from the extreme heating. This reduced the L/D at the high attitude. The corridor width and cross range are good; however, the weight of the wing and the heat protection was still too high.

Modified Lenticular Vehicle

The R-3, a modified lenticular configuration, was derived by reducing the hypersonic L/D and utilizing a more efficient elliptical structure. (Figure XVI-3-1, right.) This sacrifices some corridor width and cross-range maneuverability. It does, however, reduce the total re-entry heating and weight. The folding fins of the R-3 not only increase the L/D for sub-orbital maneuvering, but also permit a conventional dead-stick landing. Thus, preliminary optimization of the controlling factors and guided by parametric studies, a compact, light in weight, landable, re-entry vehicle was evolved.

SELECTED GLIDE CONFIGURATION

The R-3 modified lenticular re-entry vehicle with its conical transition section is shown in Figure XVI-3-3. Vertical fins fold inward, thus protecting them from the re-entry heating at high angles of attack and fold outward to provide longitudinal trim at low angle of attack, thus providing a higher L/D for maneuvering and landing.

Two solid-propellant abort rockets are attached to the back of the re-entry vehicle. They provide for abort from on-the-pad through high 'q' to orbital velocity. The conical transition section contains the mission module and the propulsion section.

The re-entry vehicle is 16 feet long, 12 feet wide, and 6 feet high. The nose radius is 12 inches and the bottom surface is a 400-inch radius sphere. The entire body is protected from re-entry by double-wall construction. The lower surface is protected additionally by a layer of ablative material. Preliminary study calls for nose skid and two main skid gears extended prior to the conventional landing.

The transition section, when attached to the re-entry vehicle during the lunar mission, brings the total vehicle to 29 feet in length.

The entry hatch is in the top of the re-entry vehicle. Figure XVI-3-4 shows the crew members sit three abreast in tiltable seats.

The double-wall section in front of the windshield can be raised to permit forward vision during the space mission. This double-wall section must be secured in place during re-entry; however, it is jettisoned to permit the pilot full forward visibility during the gliding approach and conventional landing.

The three-axes vehicle controllers are provided on the armrests on the seats. The electronic equipment is located on the right side of the cockpit. Batteries and inverters are located in the nose. (Figure XVI-3-5.) Environmental control equipment together with the fluid supplies are located on the left side of the cabin. The spherical tanks contain appropriate supplies for pressurization, and water for cooling

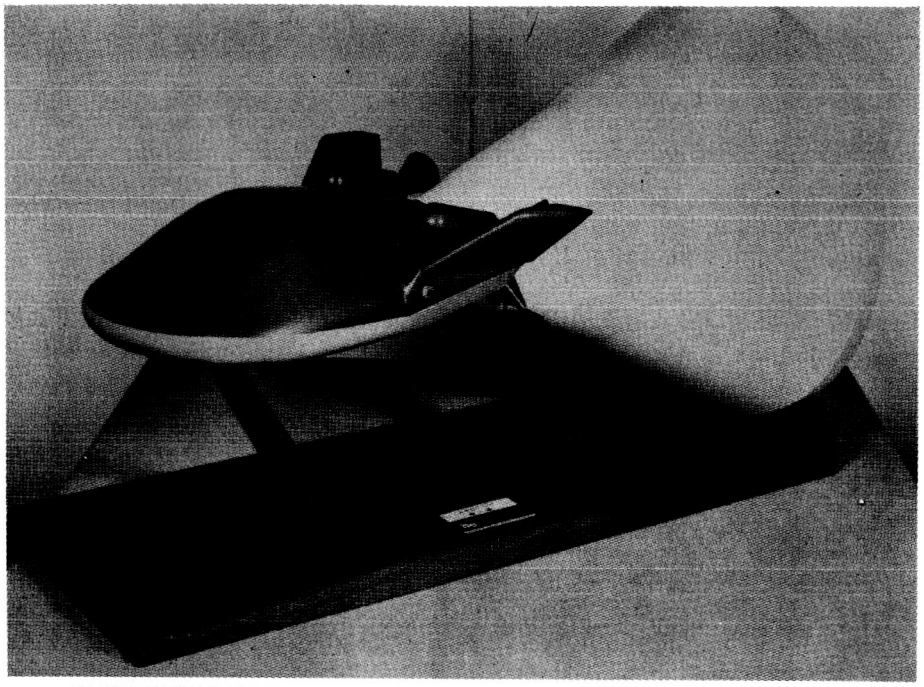


Figure XVI-3-3. Modified lenticular R-3 configuration (1/10th scale model)

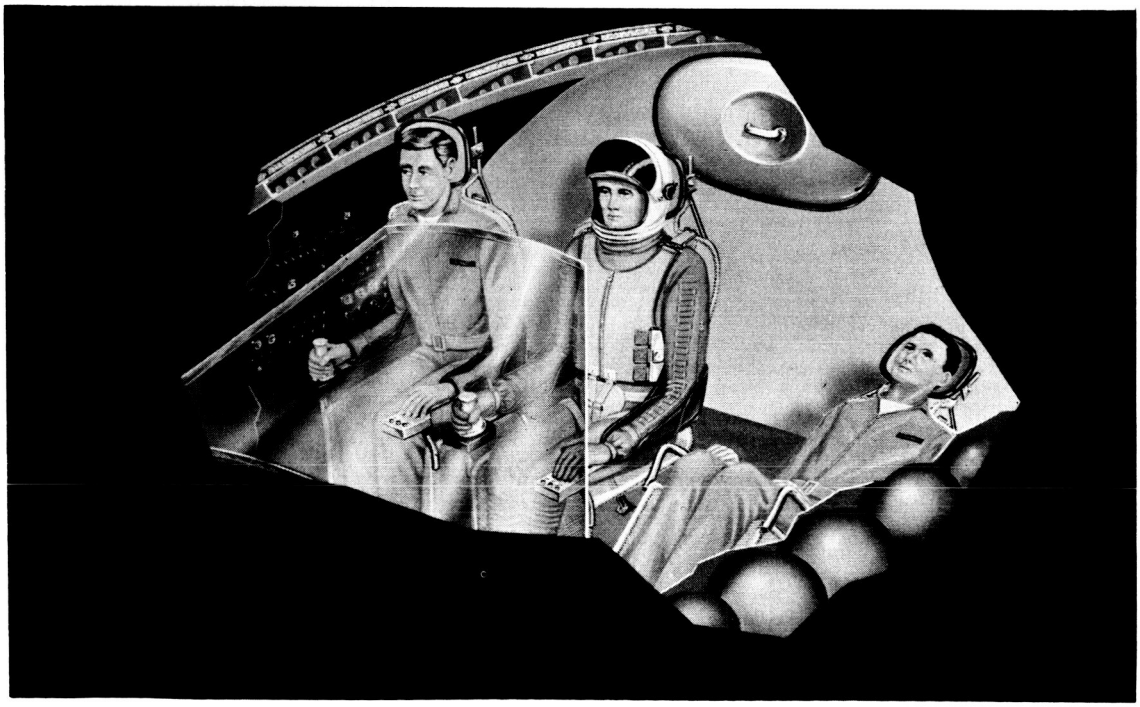


Figure XVI-3-4. Cockpit interior of modified lenticular configuration

and life support. In the rear of the vehicle are pressurized spherical tanks containing methane gas and oxygen gas for the reaction control system. This system permits instantaneous operation under zero-g conditions.

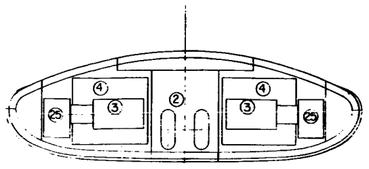
A tunnel having a cross section of 20 inches by 40 inches provides access to the mission module. Pressure hatches at both ends of this tunnel permit its use as an airlock in the event of decompression in either the control module or the mission module. It also permits access to various equipments and controls for in-flight maintenance.

Figure XVI-3-6 shows the mission module containing photographic equipment, reconnaissance instrumentation, life-support equipment, food supplies, waste disposal, and power supply for the entire vehicle. All equipment is accessible to facilitate in-flight maintenance. Access to the wall of the pressure capsule is provided to repair leaks during the mission. Windows equipped with light shields and filters permit external observation, astro-navigation, and photography.

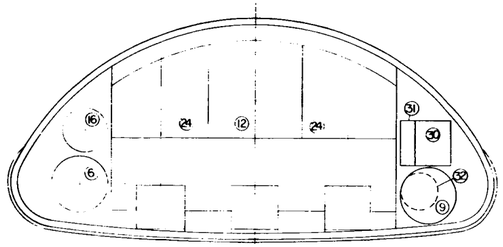
The propulsion system is located in the 30 degree conical section behind the mission module. (Figure XVI-3-7.) Two 5-foot diameter spherical tanks contain liquid hydrogen and two smaller tanks contain liquid fluorine. The hydrogen tanks are constructed from 6AI-4V heat treated titanium and insulated with Linde Company SI-4 material. The fluorine tanks are constructed of PH15-7 Mo. heat-treated stainless steel and also insulated with SI-4. Two separate tanks (uninsulated) are used to obtain pressurized gas which is used in conjunction with four 30-pound thrust chambers for fuel settling and attitude control during the mission. A recirculating propellant gas system is used to drive turbine pumps which provide high-pressure propellants to the main thrust chambers.

The normal sequence during the lunar mission is as follows: After reaching escape velocity and the proper trajectory to the Moon, the space vehicle separates from the S-IV stage of Saturn at the 14-foot diameter section. During lunar orbital work and for the return to the Earth, this configuration remains intact. Prior to re-entry, the re-entry vehicle is separated from the conical mission module by four gas-pressure pistons.

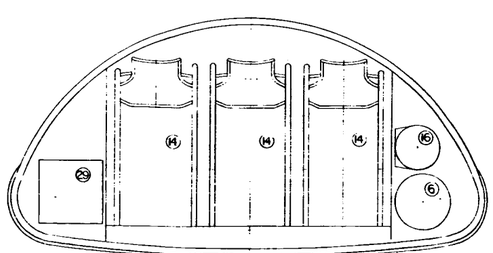
Figure XVI-3-8 shows a three-view of the modified lenticular configuration.



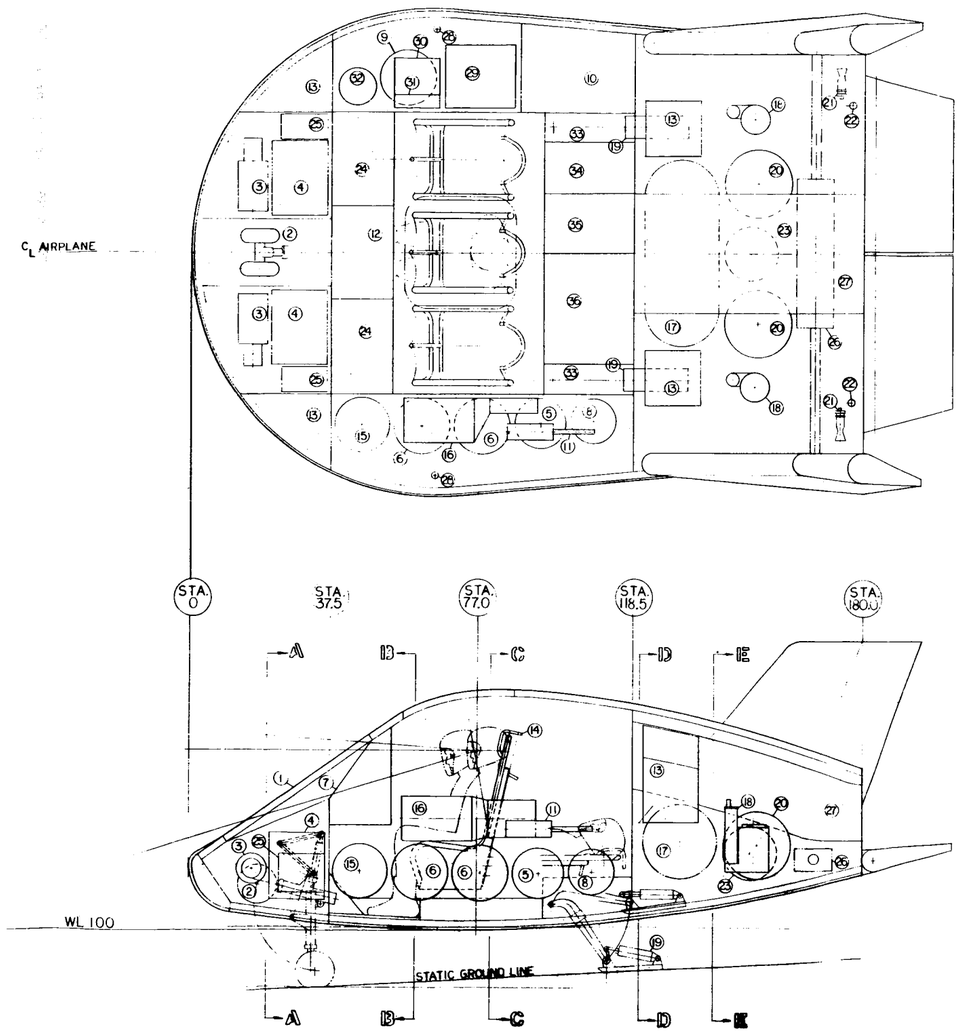
SECTION A-A

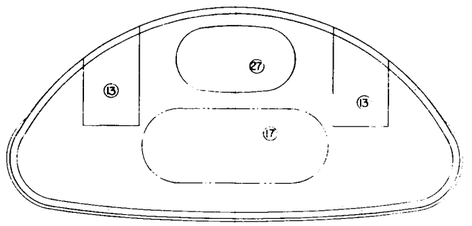


SECTION B-B

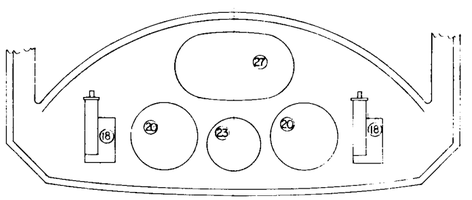


SECTION C-C





SECTION D-D



SECTION E-E

1. WINDSHIELD HEAT SHIELD (RETRACTABLE)
2. NOSE LANDING GEAR
3. HYDRAULIC PUMPS AND MOTORS
4. BATTERIES
5. NITROGEN TANK
6. OXYGEN TANK - AIR CONDITIONER
7. WINDSHIELD
8. HELIUM TANK - AIR CONDITIONER
9. EMERGENCY OXYGEN TANK
10. FOOD AND DRINKING WATER
11. EJECTOR - AIR CONDITIONER
12. INSTRUMENT PANEL
13. PARACHUTES (4)
14. CREW SEATS
15. WATER TANK - AIR CONDITIONING
16. FAN, FILTER AND HEAT EXCHANGER - AIR CONDITIONER & EQUIPMENT COOLING
17. WATER TANKS - EQUIPMENT AND STRUCTURE COOLING (SPACE RESERVED)
18. HYDRAULIC RESERVOIR AND ACCUMULATOR (3)
19. MAIN LANDING SKID GEAR
20. GASEOUS OXYGEN TANKS - REACTION CONTROLS (3)
21. REACTION CONTROL THRUSTERS - YAW
22. REACTION CONTROL THRUSTERS - PITCH
23. METHANE TANK - REACTION CONTROLS
24. EQUIPMENT COMPARTMENT
    - ELECTRICAL
    - GUIDANCE
    - COMMUNICATIONS
    - TELEMETRY
    - ELECTRONICS
25. INVERTERS
26. HEAT EXCHANGER - STRUCTURE COOLING (SPACE RESERVED)
27. TUNNEL
28. REACTION CONTROL THRUSTERS - ROLL
29. COMPUTER
30. STABLE PLATFORM
31. PLATFORM ELECTRONICS
32. EMERGENCY NITROGEN TANK
33. MAIN SKID WELL
34. PERSONAL EQUIPMENT
35. SURVIVAL EQUIPMENT
36. SECONDARY PRESSURE PROTECTION - SUIT STORAGE

Figure XVI-3-5. Inboard profile, R-3 modified lenticular configuration

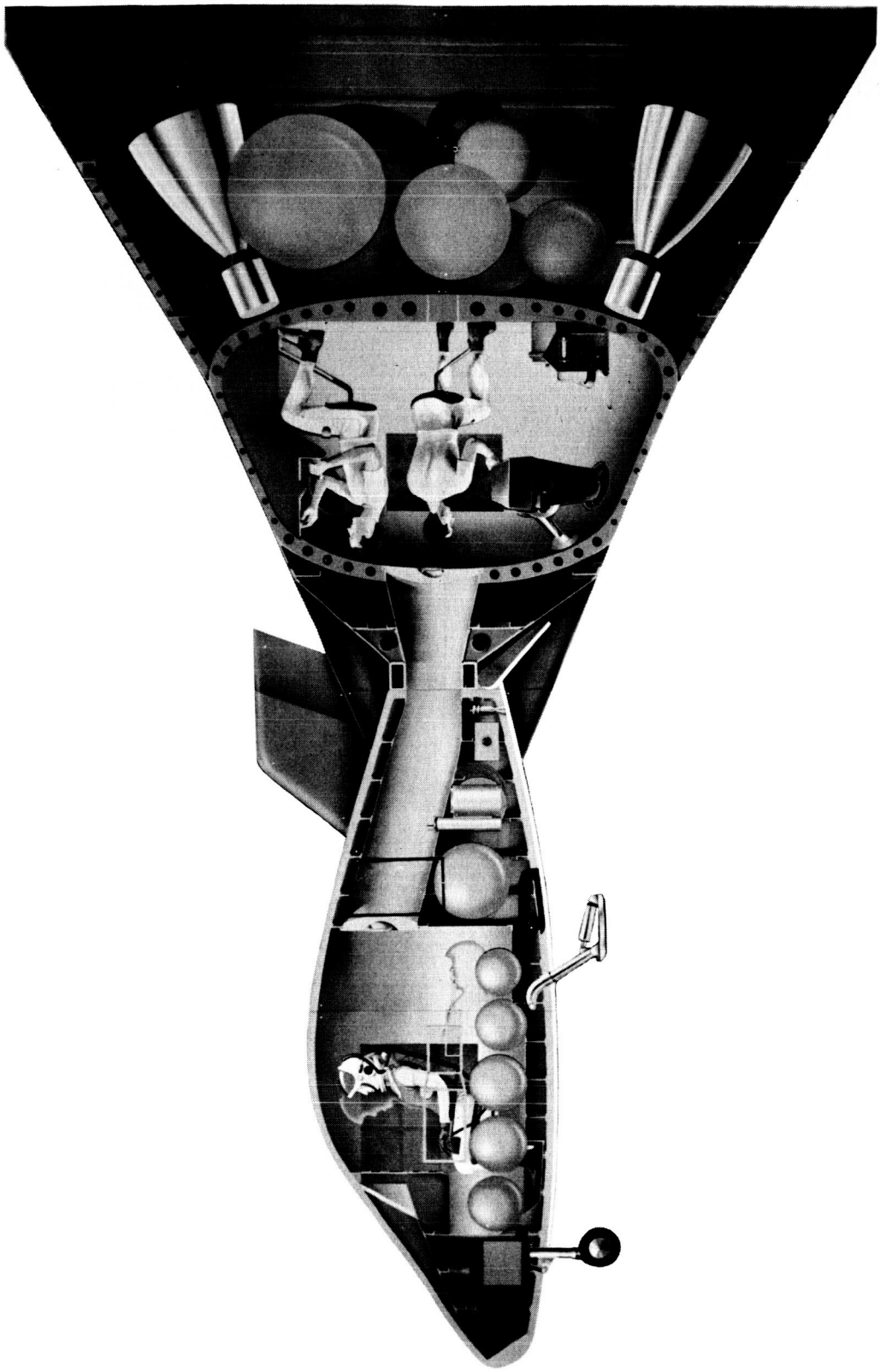
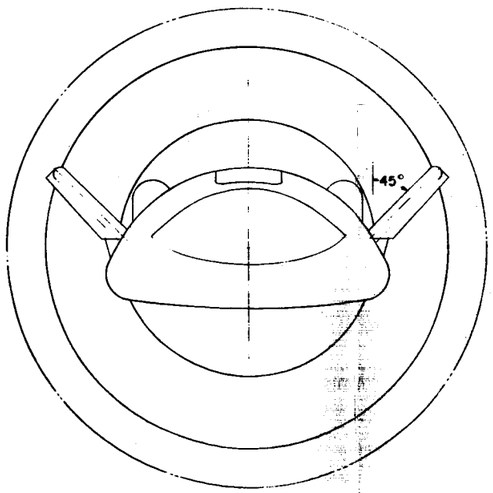


Figure XVI-3-6. General arrangement, modified lenticular configuration



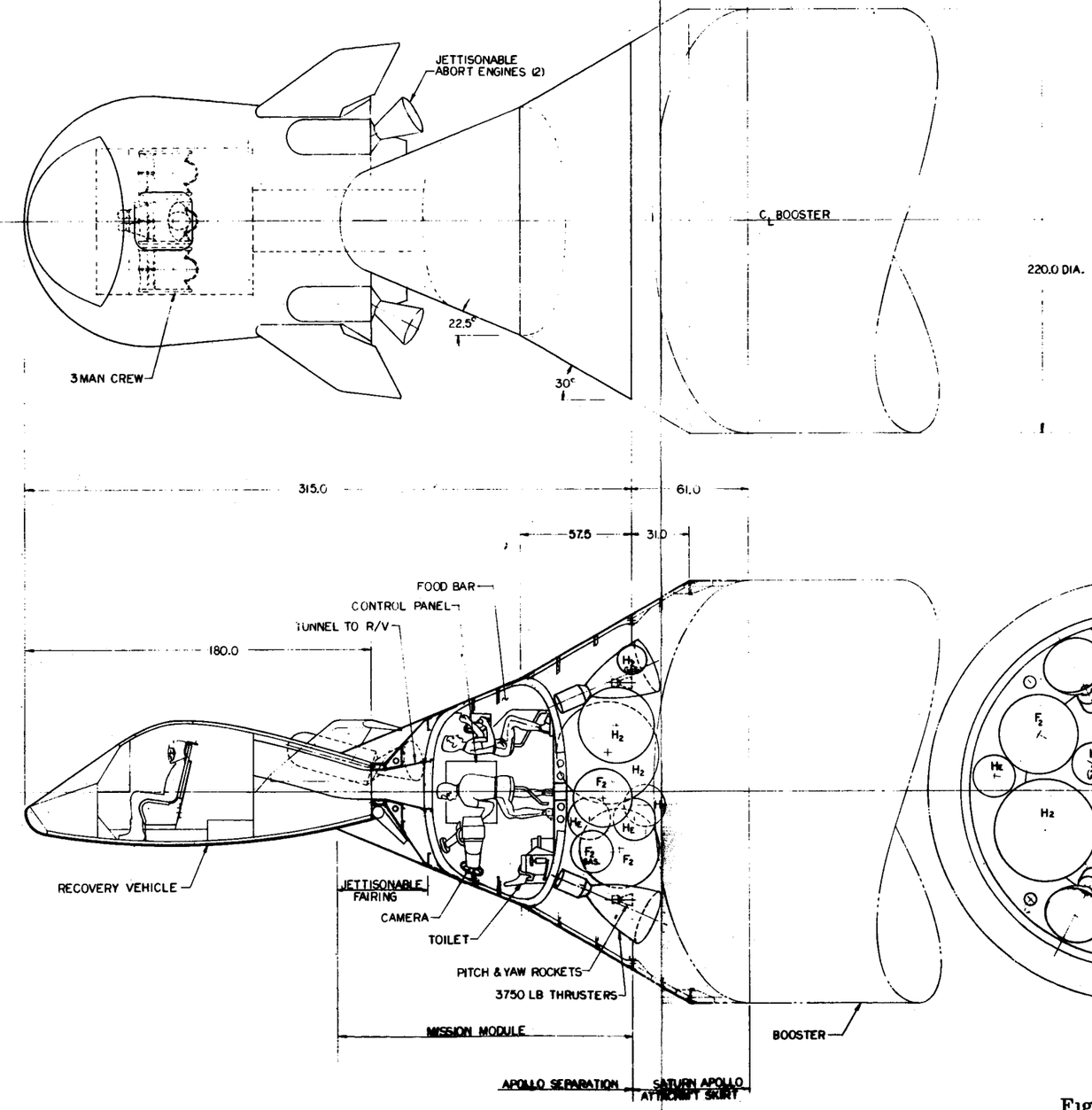


Figure XVI-3-7. General arrangement outline, modified lenticular configuration

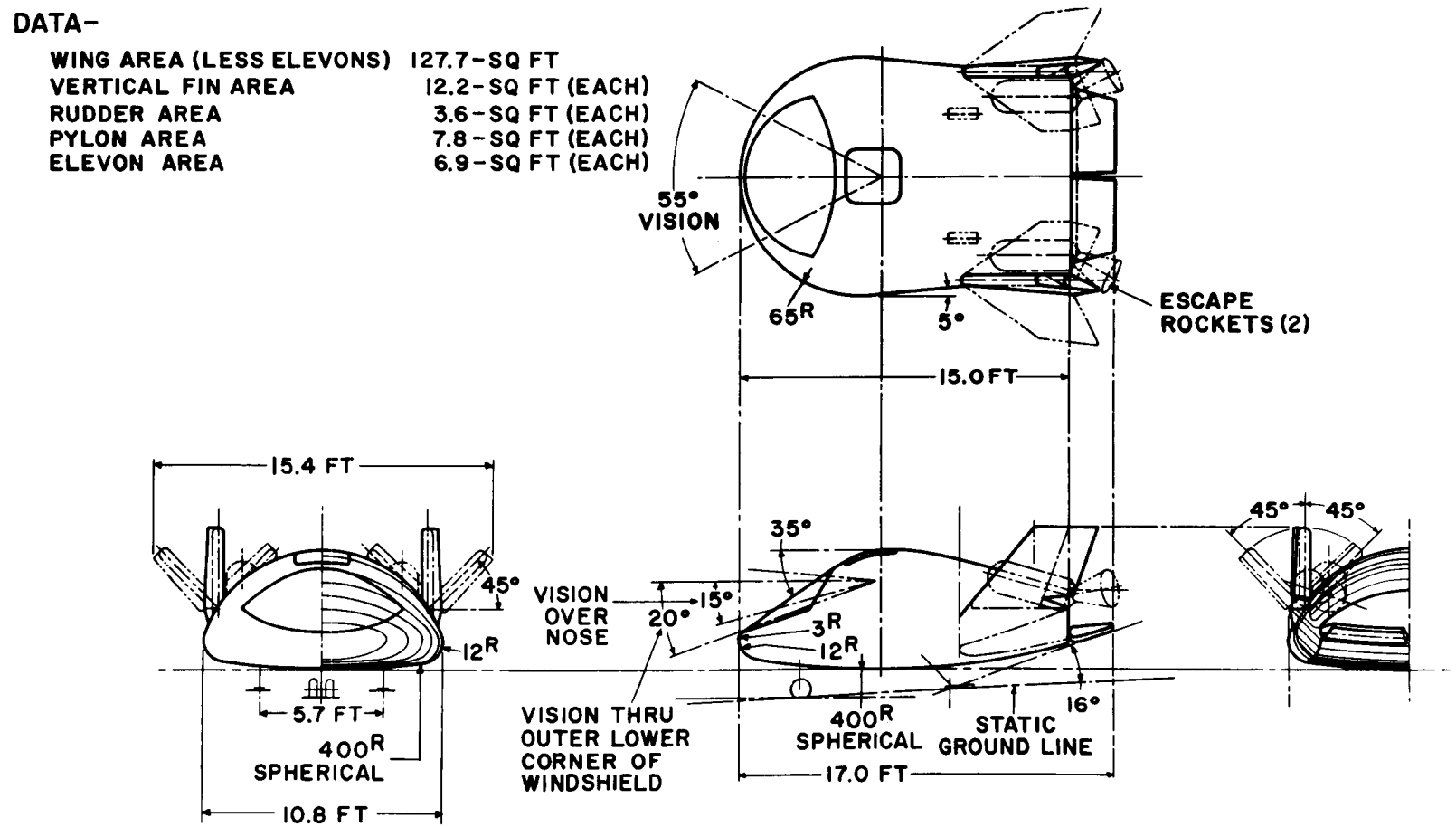


Figure XVI-3-8. Three-view of modified lenticular configuration

The modified lenticular shape is intended to provide a compact, minimum lifting surface, minimum volume configuration. It will re-enter hypersonically at attitudes above that for $C_{L_{max}}$ using a controllable range of L/D between 0 and 0.7 to widen its re-entry angle corridor. Additional range corrections can be obtained by rolling the vehicle about the wind vector to produce desired net side-range or down-range corrections. At lower hypersonic or supersonic speeds, the vehicle may make use of the higher L/D ratios available at attitudes below that for $C_{L_{max}}$.

The vertical fins are folded on top of the vehicle during the high-attitude superorbital portion of re-entry to avoid overheating these surfaces. In this configuration, directional stability is obtained from the large radius of curvature of the leading edge and from the aft sides of the body which are skewed with respect to the body axes so that they will be effective in the lateral direction. At sub-orbital speeds when the danger of overheating is past, the fins will be unfolded to a near vertical position to provide additional directional stability or tilted out even further to provide added longitudinal stability and trim capability as well. The (nominal) rudder surfaces can be used to supplement the elevon control capability when the fins are in the tilted-out positions.

The primary structure of the re-entry vehicle is conventional construction of stringers, longerons, frames and skin, using standard aluminum alloys. The pressure shell is made of easily weldable 6061 aluminum. The external-heat-protection system consists of an outer ablative coating to resist the short duration high-intensity heating. The main heat protection is radiative utilizing double-wall construction which consists of super-alloy sandwich panels backed up by packaged insulation. Weight summaries of this and improved systems using pressure-supported structure and beryllium material are included in the following sections. The construction provides, by virtue of its mass, solar flare radiation protection for the crew during the mission. Micrometeorite impact protection is shown in Figure XVI-3-25.

GLIDE VEHICLE STRUCTURE AND THERMAL PROTECTION MATERIALS

General

Structures work conducted to date for the glide vehicle study has been concerned principally with the re-entry vehicle, since this is considered a more critical structural problem than the mission module due to the aerodynamic heating problems of re-entry. The work has had two principal objectives: First, to provide structural concepts, structural arrangements and airframe weight data in support of configuration studies; and second, to investigate various approaches to the more critical problem areas. The problem areas which have been emphasized are those of airframe weight and aerodynamic heating during re-entry. Preliminary consideration has also been given to the micrometeorite problem and the problem of pressure retention for long periods of time with negligible losses.

With respect to re-entry heating, the protected structure concept was selected at the beginning of the study as the general solution. Protected construction involves a low-temperature primary structure which carries all major loads, in addition to housing the contents of the vehicle. This primary structure is protected from aerodynamic heating effects by an external system of insulation and cooling. See Figure XVI-3-9.

It is expected that the choice of protected construction will be verified later in the study by a consideration of alternate approaches. For the moment, the choice is based on past experience which includes both the development, fabrication and testing of protected structures and the study of their application to a number of hypersonic and re-entry vehicles. On the basis of this experience, it is believed that protected construction offers simple and reliable subsystems, since these all operate at low temperature. It further simplifies manufacturing problems, since 50 per cent of the airframe weight is conventional aluminum construction and the high-temperature elements are small and of simple shape. This approach also offers important growth potential in terms of higher temperature capability and reduced airframe weight since it separates strength and temperature requirements and makes possible the use of a wide range of materials. Past studies also have shown that this approach offers important weight advantages

CONFIDENTIAL

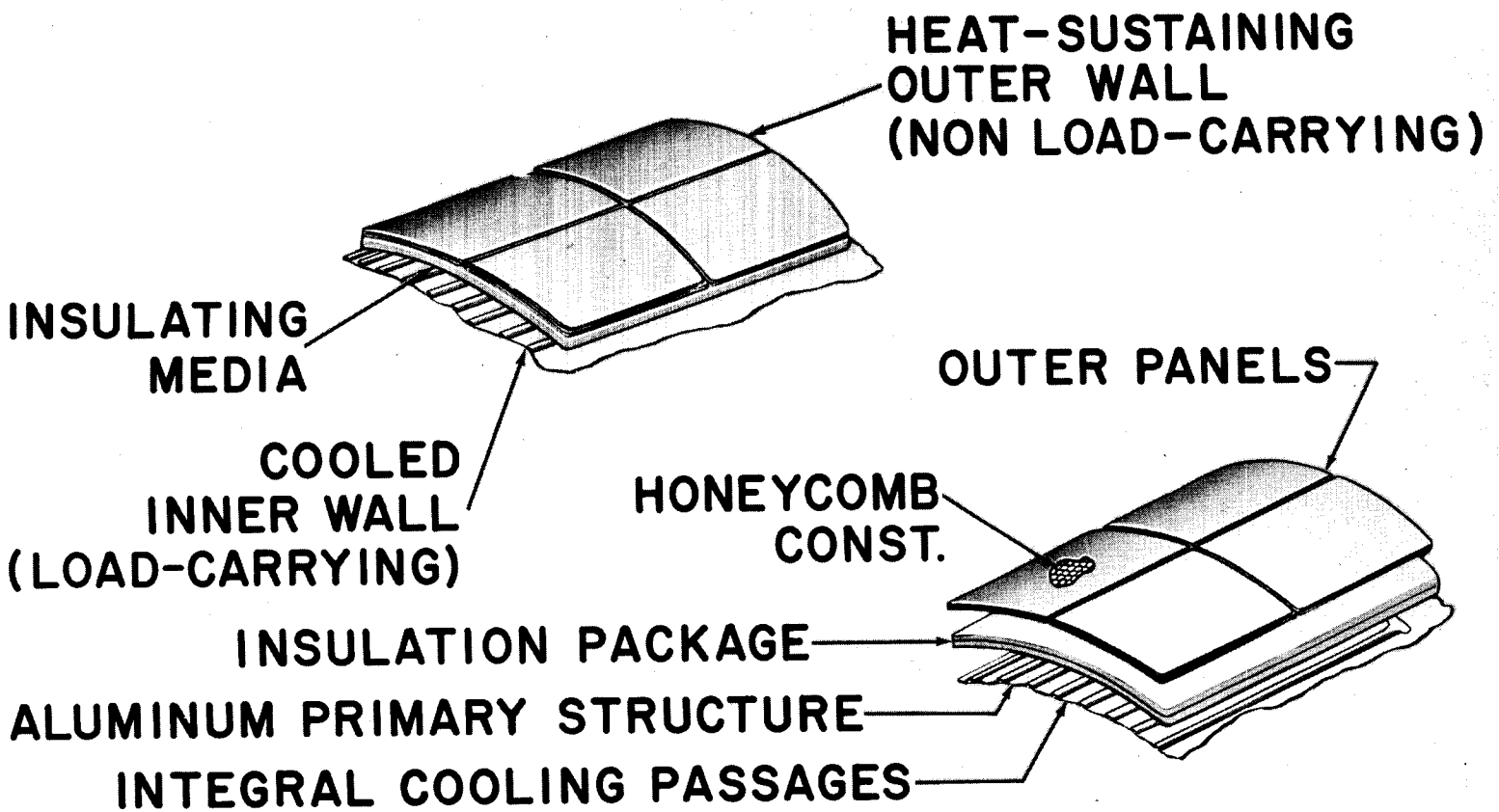


Figure XVI-3-9. Double-wall construction

CONFIDENTIAL

where a cool interior must be provided for crew, equipment, and propellants, and with an efficient vehicle configuration this condition holds for most of the airframe.

With the protected-construction approach established, the airframe study work divides into two parts; the study of the basic primary structure and the study of the protection system.

Studies of the primary structure have included the conventional construction methods of stringers, longerons, frames and skin, using the standard aluminum alloys. Consideration has also been given to a more advanced concept involving the use of pressure-supported shells. Significant weight benefits have been shown by using this concept for a high L/D wing configuration which involved a more or less cylindrical fuselage. The same approach is now being applied to the low L/D configuration but because of the shape, this represents a much more advanced application for pressure-supported structures and careful consideration is required of a number of fundamental points before feasibility is assured. Consequently, at present, firm primary structure weights for the low L/D vehicle are available only for conventional construction. One other aspect of advanced primary structures which is being considered is the introduction of beryllium. Again only tentative weight indications are available at present.

With respect to the thermal protection system the problem of lifting re-entry from escape velocities, in combination with significant maneuvering capability for landing accuracy, requires what might be termed a "wide-range" protection system. This term refers to a system which simultaneously has the temperature capability to resist very severe heating for short times, and also the protective efficiency to sustain moderate heating for long periods of time with practical weights.

Normally, protection systems based on heat absorption would be used for high intensity short-time conditions, since such systems are not particularly sensitive to heat intensity and their relatively poor efficiency is not serious for times of one to two minutes. Similarly the long-time heating condition would require the use of a radiative system, which is normally limited in the heating intensity which can be sustained, by the temperature capability of surface materials.

For the present application, which involves the combination of high intensity for short times and moderate intensity heating for long times, both of these systems have been considered but advanced forms have been assumed in order to improve the deficiencies of each. Absorptive systems have been considered with materials which have good insulative or radiative capabilities in order to improve weight efficiency. Also radiative systems have been extended considerably in temperature capability by considering nonmetallic materials and by accepting some deterioration of these materials in order to accommodate the maximum heat fluxes. In addition to these extended forms of absorptive and radiative protection systems, combinations of both have also been introduced in an effort to find the minimum weight arrangement.

In this report, the work discussed above is summarized in a number of illustrations which include primary structure weights for both high and low L/D vehicles using conventional and pressure-supported structures, and also the unit weights of a number of protection system arrangements, classified as discussed above. Total airframe weights are also summarized by using as a starting point a conventional primary structure for the low L/D vehicle in combination with a thermal protection system which also represents present day capabilities. Potential airframe weight improvements by introducing the various advanced concepts in the primary structure and the protection system are then demonstrated.

With respect to the problem of micrometeorite damage, the approach which has been used in the present study has been to examine the capability of the structure that is required for strength and resistance to aerodynamic heating, because the constructions which will be proposed are inherently suitable for resisting micrometeorite impact. Two criteria have been established to evaluate airframe resistance to micrometeorite impact: First, penetrations of the pressure shell which are large enough to be catastrophic and non-repairable before serious pressure loss should have a very low probability of occurrence, comparable with catastrophic failures in conventional military aircraft. Second, penetrations which are likely to occur more frequently than this should be detectable and repairable before serious loss of internal pressure.

Structure

The charts presented herein have been prepared to provide a basis upon which one may judge the relative merits of various configurations with respect to weights of the vehicle primary aluminum structure and the thermal protection system.

The two configurations considered to be of fundamental interest for this program are the:

(a) High L/D delta-winged glider which is similar to the Dyna-Soar vehicle.

(b) Lifting body shape referred herein as the Modified Lenticular configuration.

Accordingly, preliminary structural drawings have been prepared with sufficient detail and analysis to permit realistic weight estimates. These structural drawings are shown in Figures XVI-3-11 and XVI-3-12 and are based on the design criteria summarized in Figure XVI-3-10. Finally, the corresponding primary structure weight comparison is shown on Figure XVI-3-13. Figure XVI-3-14 is presented for the sole purpose of illustrating the advantages of pressure-supported type of structures.

Referring to Figure XVI-3-14, it is significant to note that the primary structure weights for these two configurations are very closely matched in spite of the large differences in surface areas. Actually these results should not be too surprising since almost 80 per cent of the total primary structure weight occurs in the cockpit and equipment compartment region of the modified lenticular configuration while the corresponding percentage for the more efficient shape of the pressure supporting shell of the high L/D vehicle is less than 50 per cent. In the over-all picture, however, the indicated advantage is lost to configuration differences since the high L/D vehicle requires wing surfaces and somewhat larger areas for both the elevon and tail.

With these observations in mind, one can conclude with some optimism that weight advantages can be similarly extended to the lenticular configuration by exploiting the concept of pressure supported structures. As a matter of fact, since wings are not generally employed with these lifting-body configurations, the concept is ideally suited to designs belonging to the class of pressurized shells. In particular, since the planform

CONFIDENTIAL

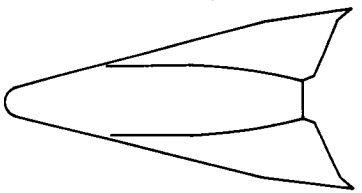
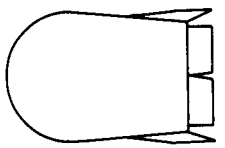
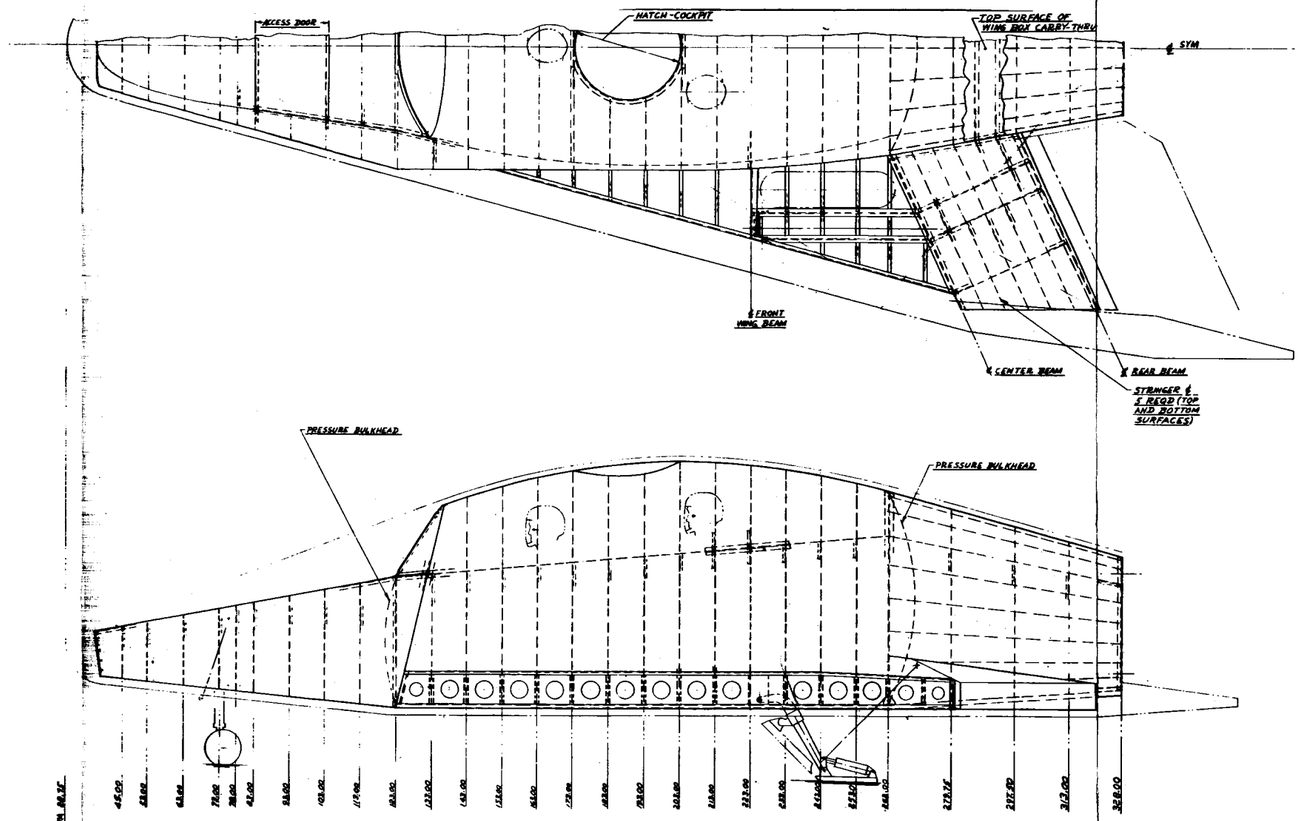
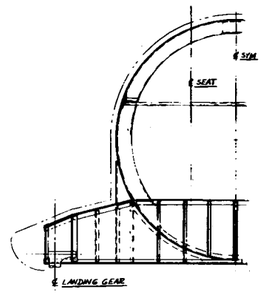
CRITICAL DESIGN CONDITIONS	HIGH L/D (R-1) 	MODIFIED LENTICULAR (R-3) 
1.- q_β -MAX. DURING BOOST V = 1256.6 FT/SEC h = 32,810 FEET α = 5 DEGREES q_β = 597 LBS/SQ FT	AFT FUSELAGE AND TAIL	AFT FUSELAGE AND TAIL
2.- INTERNAL PRESSURE p = $2p_w$ = 16 P.S.I. (BURST)	CABIN	CABIN
3.- RE-ENTRY MANEUVER	WINGS (7.33 g LIMIT)	— — — — — — —
4.- LANDING	BODY NOSE	BODY NOSE

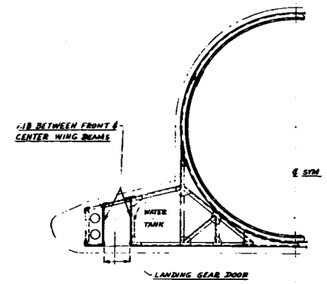
Figure XVI-3-10. Primary structure design criteria

CONFIDENTIAL

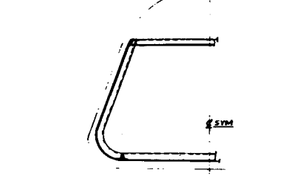




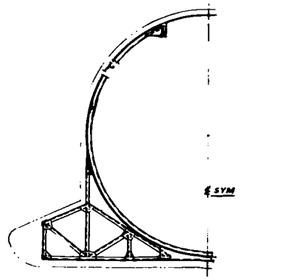
STA 223.00



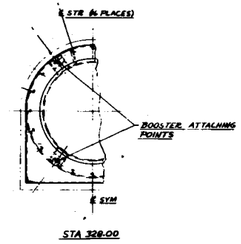
STA 243.00 SHOWN
(TYP FOR STA 233.00 & 253.00)



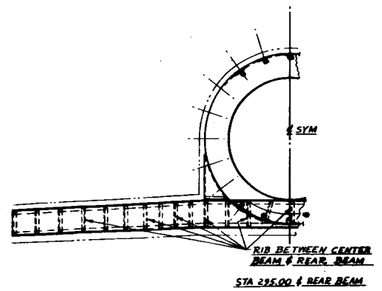
STA 113.00 SHOWN
TYPICAL FOR STA 65.00, 53.00, 63.00, 73.00, 83.00, 93.00 & 103.00



STA 123.00 SHOWN
TYPICAL FOR STA 133.00 THRU STA 213.00 INCL.



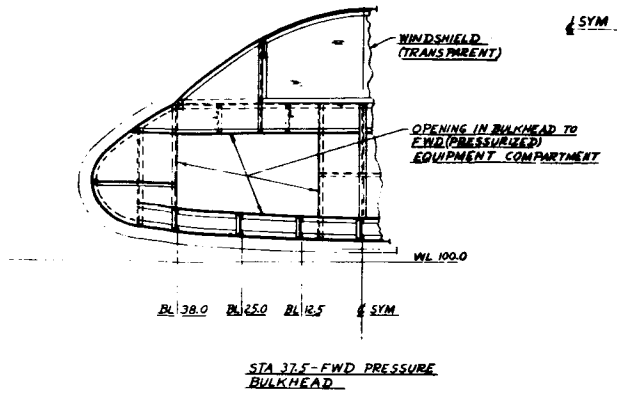
STA 328.00



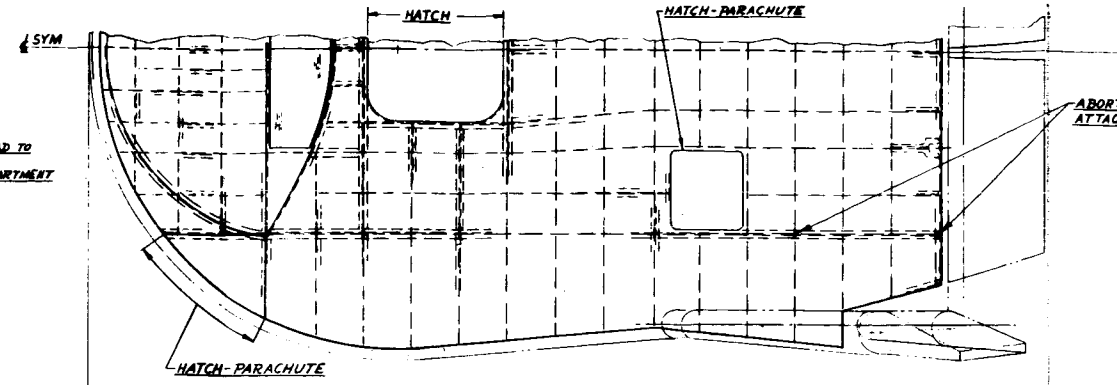
STA 295.00 & REAR BEAM

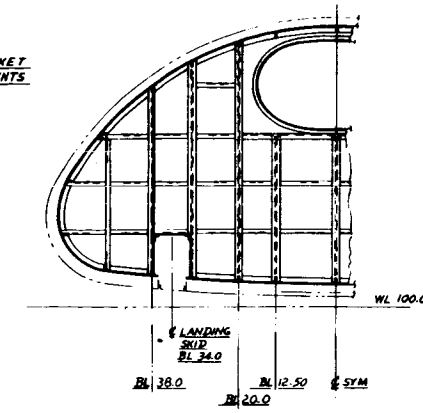
Figure XVI-3-11. Structural arrangement, R-1 high L/D glider

[redacted]

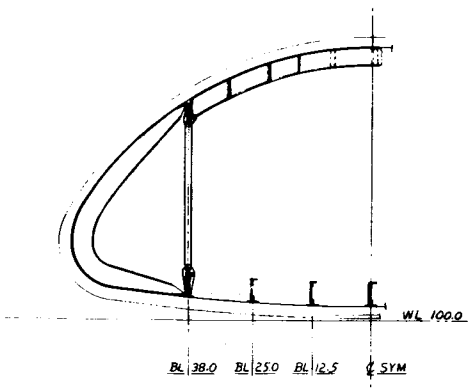


STA 37.5 - FWD PRESSURE BULKHEAD

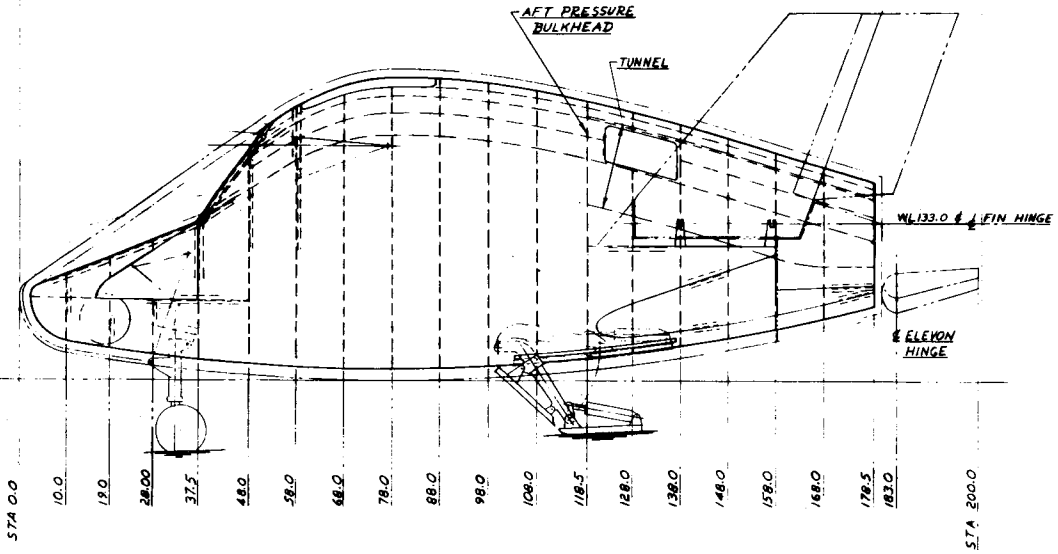


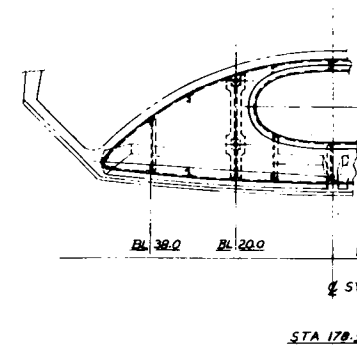


STA 118.5 - AFT PRESSURE BULKHEAD

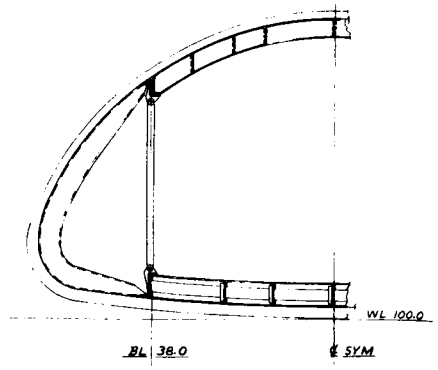


STA 58.0 TYP FOR STA 48.0

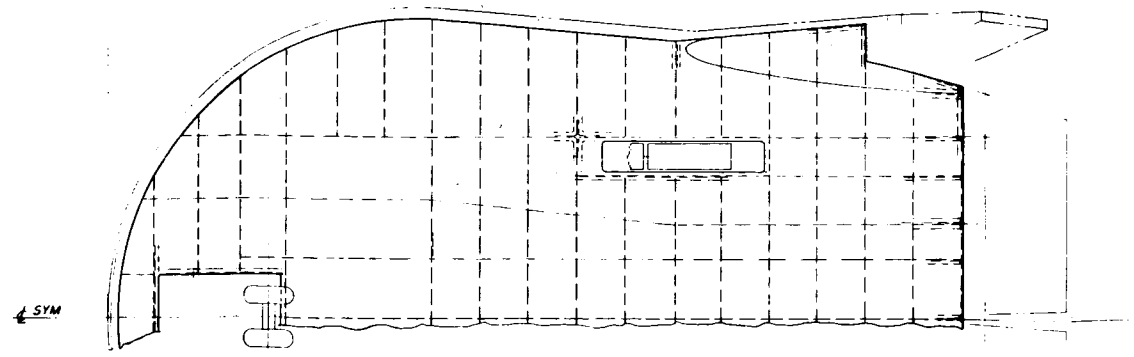


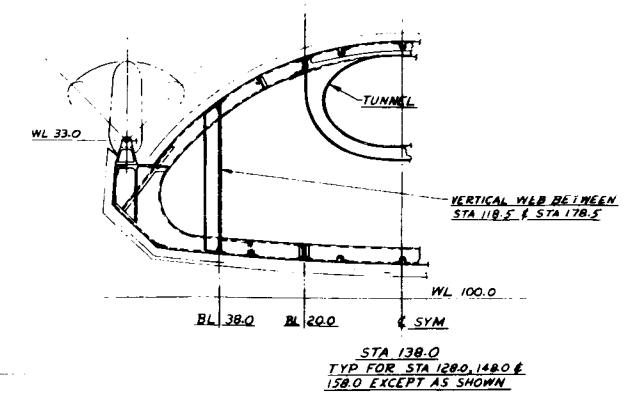


STA 178.50



STA 88.0 TYP FOR STA 78.0





STA 138.0 TYP FOR STA 128.0, 148.0 & 158.0 EXCEPT AS SHOWN

Figure XVI-3-12. Structural arrangement, R-3 modified lenticular

[redacted]

ITEM \ CONFIGURATION	PRESSURE SUPPORTED HIGH L/D VEHICLE (R-1)			MODIFIED LENTICULAR (R-3)		
	UNIT WEIGHT (LBS/SQ FT)	AREA (SQ FT)	TOTAL WEIGHT (LBS)	UNIT WEIGHT	AREA	TOTAL WEIGHT
WING	2.00 *	107.0 *	214	0	0	0
ELEVON	2.00 *	48.5 *	97	2.00	30.0	60
TAIL	2.55 *	81.2 *	207	2.55	40.0	102
BODY-NOSE	2.00	79.5	159	1.91	50.2	96
— COCKPIT	1.60	232.5	372	3.60	174.7	630
— EQUIPMENT COMPARTMENT	2.70	85.1	230	4.00	106.3	425
TOTAL	2.02	633.8	1279	3.28	401.2	1313

Figure XVI-3-13. Primary structure weight comparison

CONFIDENTIAL

CONFIGURATION / ITEM	DYNA-SOAR TYPE			PRESSURE SUPPORTED HIGH L/D VEHICLE (R-1)		
	UNIT WEIGHT (LBS/SQ FT)	AREA (SQ FT)	TOTAL WEIGHT (LBS)	UNIT WEIGHT	AREA	TOTAL WEIGHT
WING	1.85	107.0	198	2.00 *	107.0 *	214
ELEVON	2.00	48.5	97	2.00 *	48.5 *	97
TAIL	2.55	81.2	207	2.55 *	81.2 *	207
BODY - NOSE	2.50	79.5	199	2.00	79.5	159
- COCKPIT	2.70	232.5	628	1.60	232.5	372
- EQUIPMENT COMPARTMENT	2.70	85.1	230	2.70	85.1	230
TOTAL	2.46	633.8	1559	2.02	633.8	1279

* WETTED AREA

Figure XVI-3-14. Primary structure weight comparison

CONFIDENTIAL

of the lenticular vehicle can be made to fit a circular geometry the overall shape of the vehicle can be completely described by an ellipsoidal body of revolution. This feature is illustrated in Figure XVI-3-15 and more fully discussed in the subsequent Section under the heading of "Primary Structure Side Studies". These studies roughly indicate a weight saving of 0.90 lbs per sq ft, however, the advantages of this design are not restricted to weight benefits since this type of construction is visualized to be superior with respect to maintaining near-zero leakage under internal pressurization. Moreover, there is an added margin of safety against meteoroid penetration as a consequence of the heavier gages employed in fabricating to the shell concept. (With the more conventional cylindrical approach, thin gages are employed in conjunction with heavy frames which must resist all the pressure forces by beam bending action. With shells of revolution, the opposite condition prevails.)

The results of the above preliminary studies are summarized in Figure XVI-3-16 which compare the weight of the modified lenticular shape with the pressure supported vehicle having an ellipsoidal body of revolution. An approximate weight saving of 200 pounds is indicated for this design.

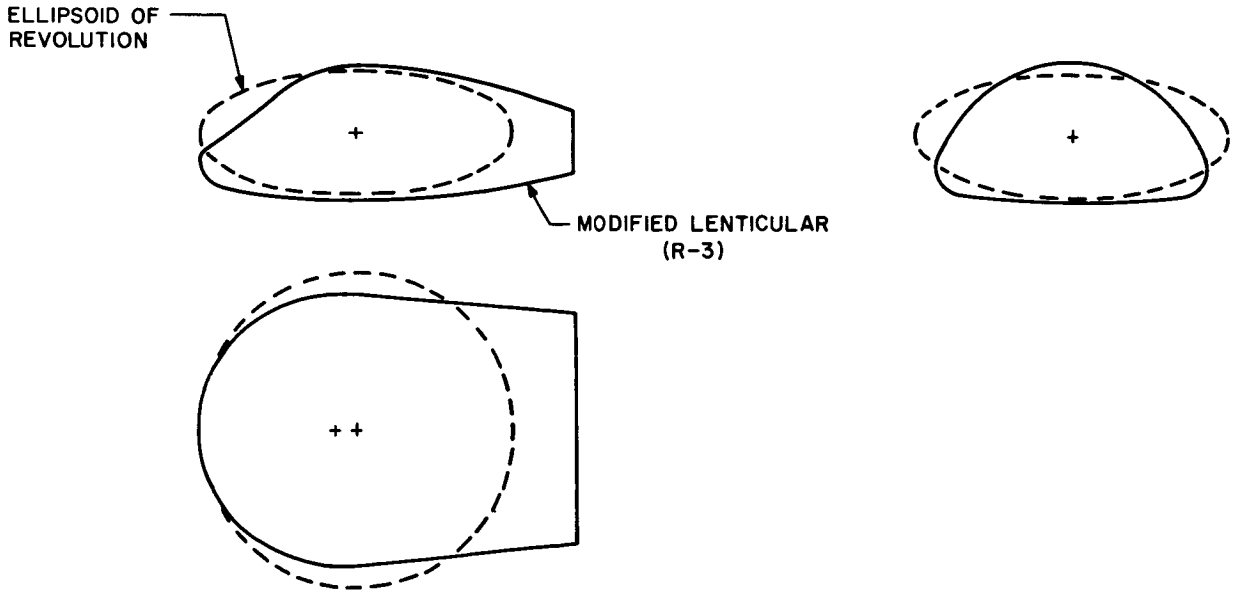


Figure XVI-3-15. Primary structure side studies; comparison of shapes

ITEM \ CONFIGURATION	MODIFIED LENTICULAR (R-3)			ELLIPSOID OF REVOLUTION		
	UNIT WEIGHT (LBS/SQ FT)	AREA (SQ FT)	TOTAL WEIGHT (LBS)	UNIT WEIGHT	AREA	TOTAL WEIGHT
WING	0	.0	0	0	0	0
ELEVON	2.00	30.0	60	2.00	30.0	60
TAIL	2.55	40.0	102	2.55	40.0	102
BODY-NOSE	1.91	50.2	96	* 2.00	313.0	626
—COCKPIT	3.60	174.7	630			
—EQUIPMENT COMPARTMENT	4.00	106.3	425	** 3.00	106.3	320
TOTAL	3.28	401.2	1313	2.25	489.3 ***	1108

* — ELLIPSOIDAL BODY
** — AFT BODY
*** — EXPOSED SURFACE MORE NEARLY MATCHES THE R-3 CONFIGURATION.

Figure XVI-3-16. Primary structure weight comparison

Figure XVI-3-25 summarizes initial studies concerned with evaluating the ability of the double-wall structure to serve as a shielding system for meteoroid protection. The information presented on these charts is based on recently published works by M. Kornhauser.

Primary Structure Side Studies

During the latter phases of this study, it became increasingly apparent that attractive weight savings can be achieved by applying the concept of pressure-supported structures to the lenticular configuration of the APOLLO vehicle. Although these studies have been limited to investigations of a general nature the results are nevertheless sufficiently encouraging to justify additional work for the next study phase.

In Figure XVI-3-17 a brief comparison study is presented in order to provide an indication of expected weight saving by fully exploiting the concept of pressure supported structures.

The study compares the basic structural weights for two types of design. Design A, Figure XVI-3-18, is an ellipsoidal body of revolution while design B is of conventional construction utilizing the combination of skin and frames in resisting effects of internal pressure.

Based on using 6061 aluminum, results indicate that Design A is lighter by approximately 0.90 lbs per sq ft.

DESIGN A: ELLIPSOID OF REVOLUTION $k = \frac{a}{b} = 2.5$

a = 75" major semi-axis

b = 30" minor semi-axis

p_w = internal working pressure = 8.0 psi

p_p = proof pressure $\cong 1.10 \times 8.0 = 8.8$ psi

p_b = ultimate or burst pressure = $2 \times 8 = 16$ psi

FOLLOWING FIXED CONDITIONS HAVE BEEN ARBITRARILY SELECTED:

a) MAXIMUM INTERNAL PRESSURE = 2(8) 16 P.S.I. (ULT.)

b) MAXIMUM VOLUME OF CABIN = 408 CU.FT.

c) MAXIMUM PLANFORM DIMENSION = 150 IN. (12.5 FT.)

d) MAXIMUM SECTION HEIGHT = 60 IN. (5.0 FT)

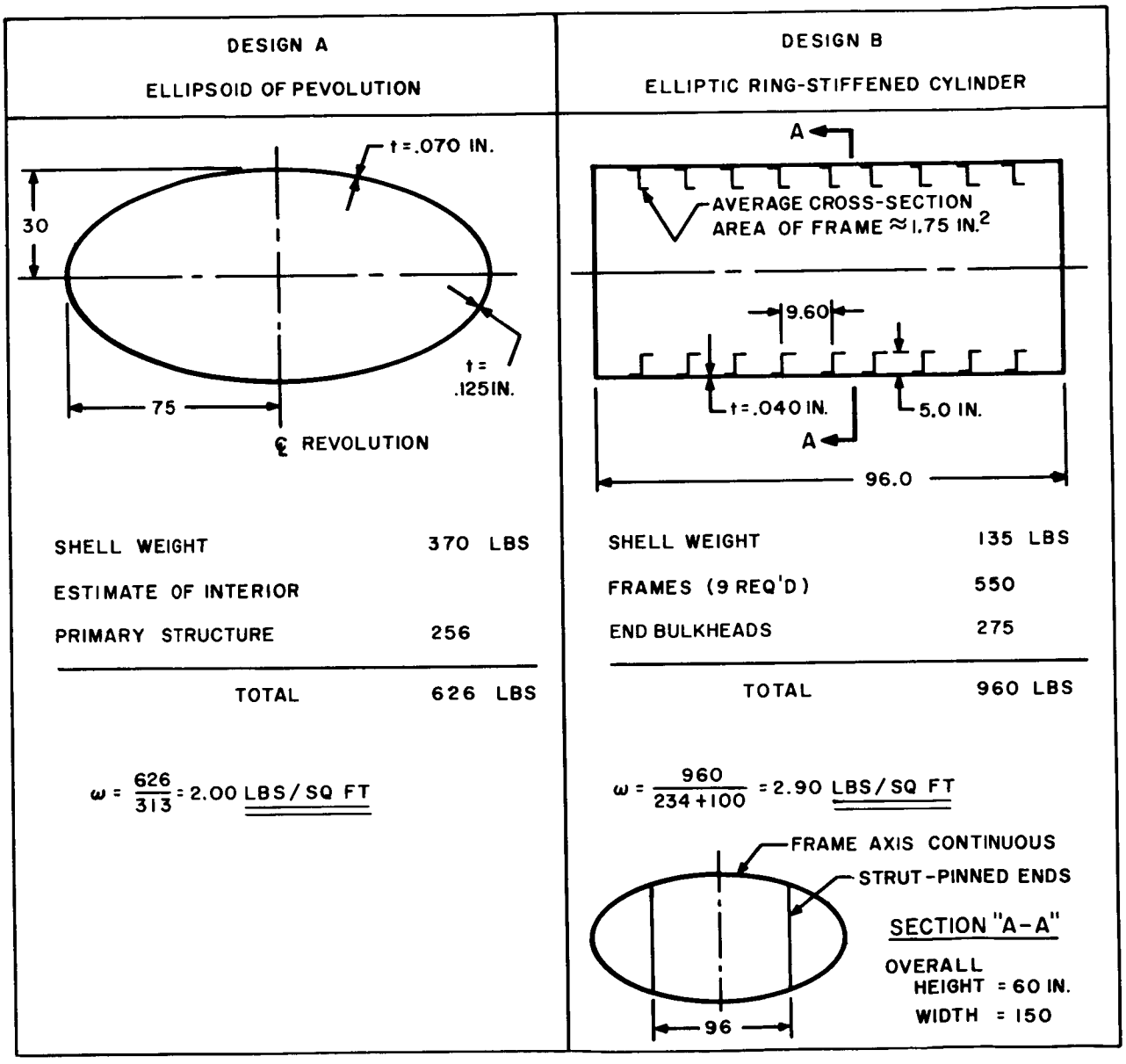


Figure XVI-3-17. Comparison of weight saving

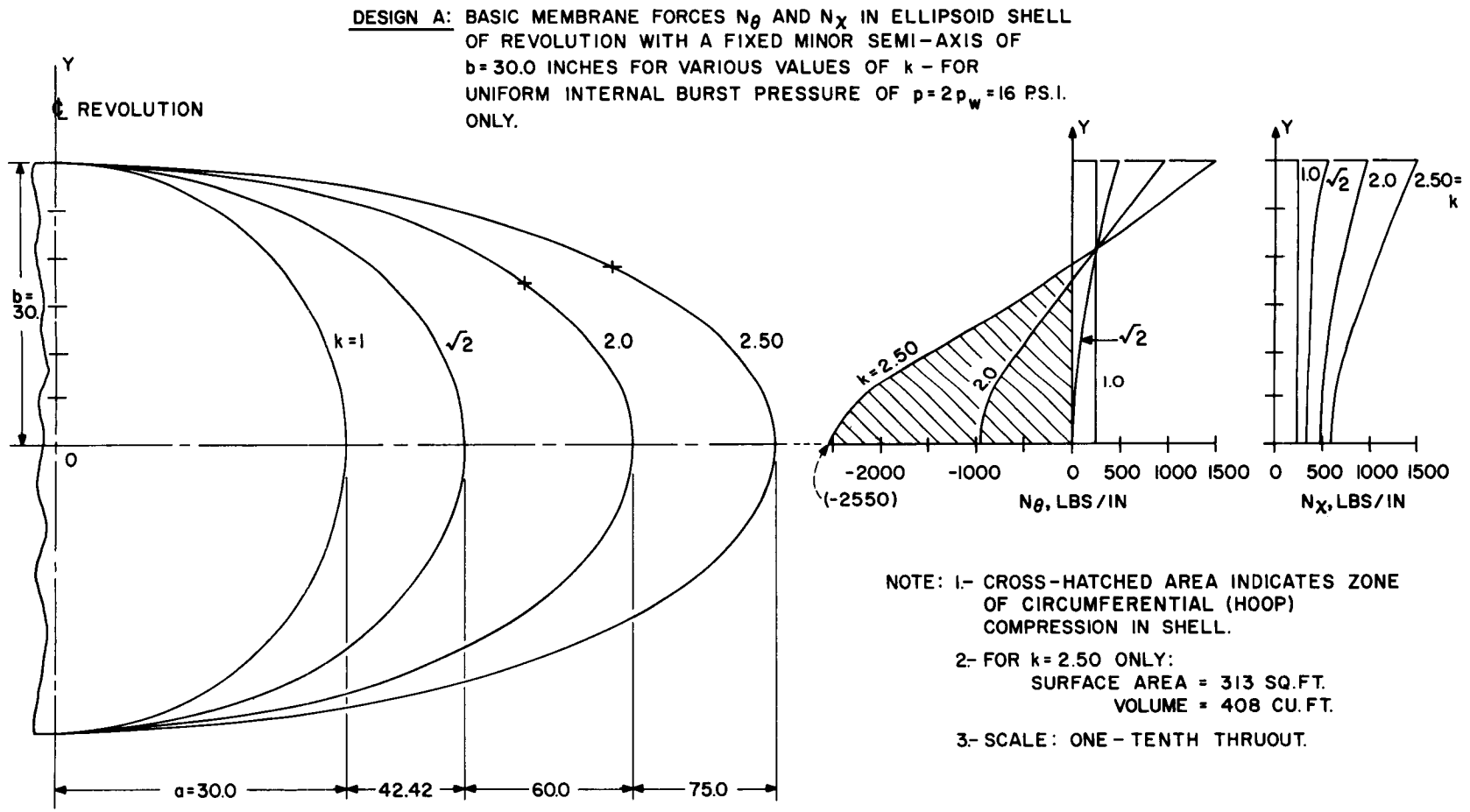


Figure XVI-3-18. Design A characteristics

The basic membrane forces in the above shell are obtained by the well-known equations:

$$N_{\varphi} = \frac{pR_2}{2} \quad \text{lbs/in}$$

$$N_{\theta} = p\left(R_2 - \frac{r_2^2}{2r_1}\right)$$

For the case of $p = 16$ psi and using the above dimensions the complete distribution of these membrane forces are summarized as follows:

In selecting the basic wall thickness for the shell, cognizance of deflection restrictions at point (1) is recognized as the controlling factor; example: At point (1):

$$r_1 = r_2 = \frac{a^2}{b} = 187.5 \text{ in.}$$

Therefore, based on proof pressure and by limiting the deflection at point (1) to, say, 0.50 inches, the required wall thickness is 0.070 inches. The corresponding maximum membrane stress is 1500/0.070 = 21,000 psi, as compared with material allowable strength of 38,000 psi.

The above deflection can be easily obtained by methods presented by W. M. Coates in ASME Transactions, 1930.

At the ends of the major axis, large hoop compression forces of an amount equal to -2550 lbs/in. exist. Since a compression zone exists, the wall thickness for this zone is obtained by equating the allowable buckling stress of a cylinder under axial compression:

$$\sigma_{CR} = \frac{0.2\,Et}{R} = \frac{2550}{t}$$

$$\therefore\ t \approx 0.125 \text{ in.}$$

For weight purposes, this thickness is conservatively assumed to extend 18.0 inches into each quadrant measured from ends of major axis.

DESIGN B: ELLIPTIC RING-STIFFENED CYLINDER ($k = \frac{a}{b} = 2.50$)

For purposes of initial weight estimates the cabin cross-section will be approximated by an ellipse for its shape. For internal ultimate pressure of 16 psi and frames spaced at approximately 10.0 inches apart, the shell wall thickness is 0.040 inch, 6061 aluminum.

Initial study of an elliptical frame, with $k = 2.50$, subjected to a uniform loading of $10(16) = 160$ lbs per inch shows that large bending moments exist at the ends of minor and major axes of an amount of +147,200 in-lbs and -225,500 in-lbs, respectively. When two interior struts are incorporated, substantial reduction in bending at the ends of minor and major axes is experienced: +64,700 in-lbs and +31,700 in-lbs, respectively; at point of strut attachment the bending moment is -124,300 in-lbs. Therefore, on the basis of utilizing two interior struts, the average cross-sectional area of the frame is 1.75 square inches and the weight of a single frame is 61 pounds, 6061 aluminum construction.

Thermal Protection

The work on thermal protection has been concentrated in the area of protection for the most severely heated regions of the R-3 modified lenticular vehicle. The thermal protection systems evaluated and be divided into three groups: all ablation, ablation with radiation, and all "radiation", as shown on the thermal protection weight summary. The heating information on which all systems are based is given in the heat flux versus time curves and the total heat per unit area versus time curves, Figures XVI-3-19 and XVI-3-20 in which the 10 g limit and the skip limit represent the range of operation over which the thermal protection must perform. Heat flux and total heat values are predicated on a 5.5-foot radius vehicle planform, and are corrected by a factor of 0.7 for the similarity of the lower surface to a disc rather than a hemisphere.

The ablative material used for some of the systems described here was Teflon, which should be regarded as a class of materials rather than a particular material; it was chosen primarily because its behavior is reasonably well understood, it has a low ablation temperature, and its value of heat of ablation is relatively high. The heat of ablation used in determining the weights of ablative material required was the value

CONFIDENTIAL

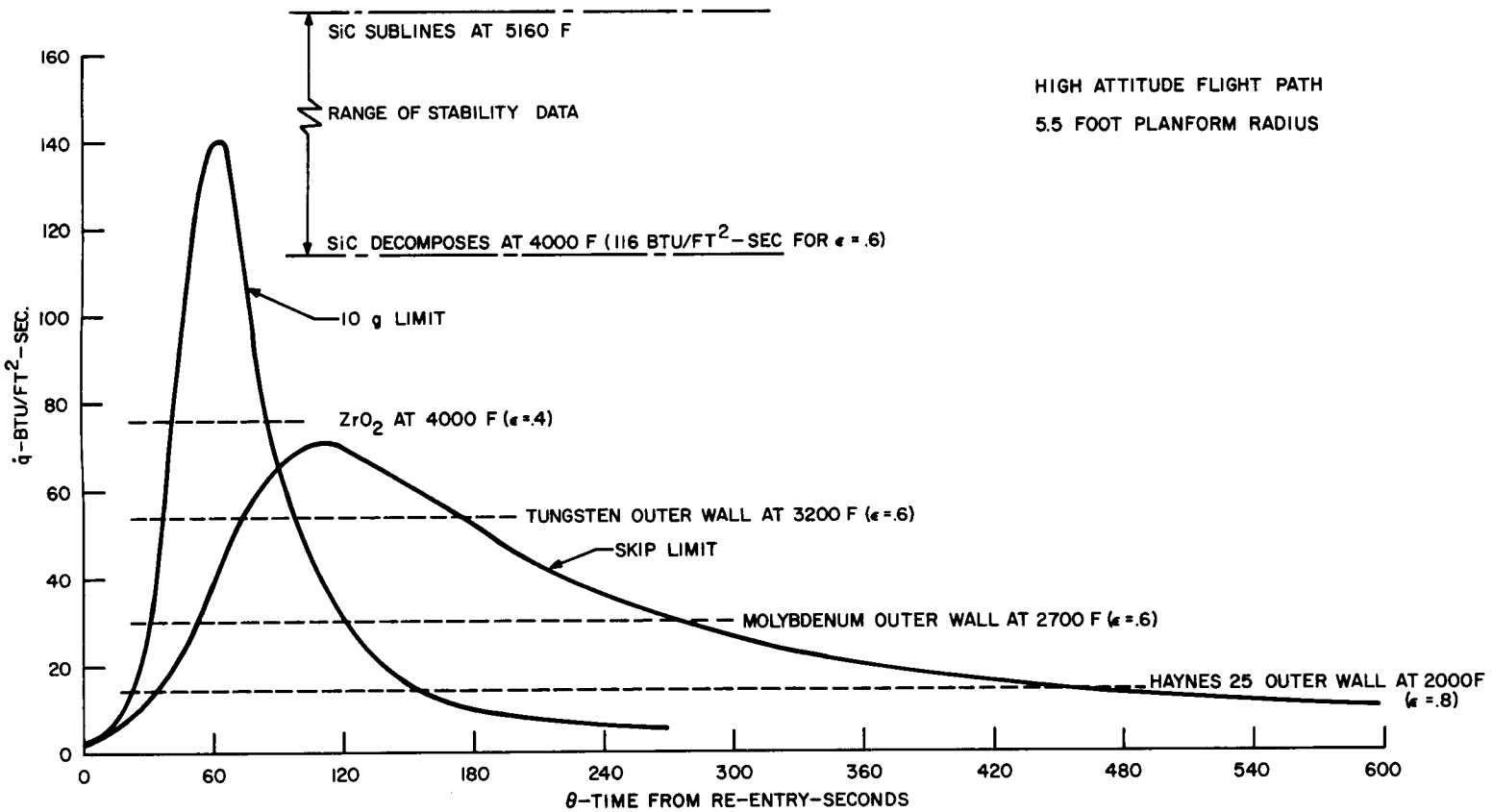


Figure XVI-3-19. Heat flux versus time

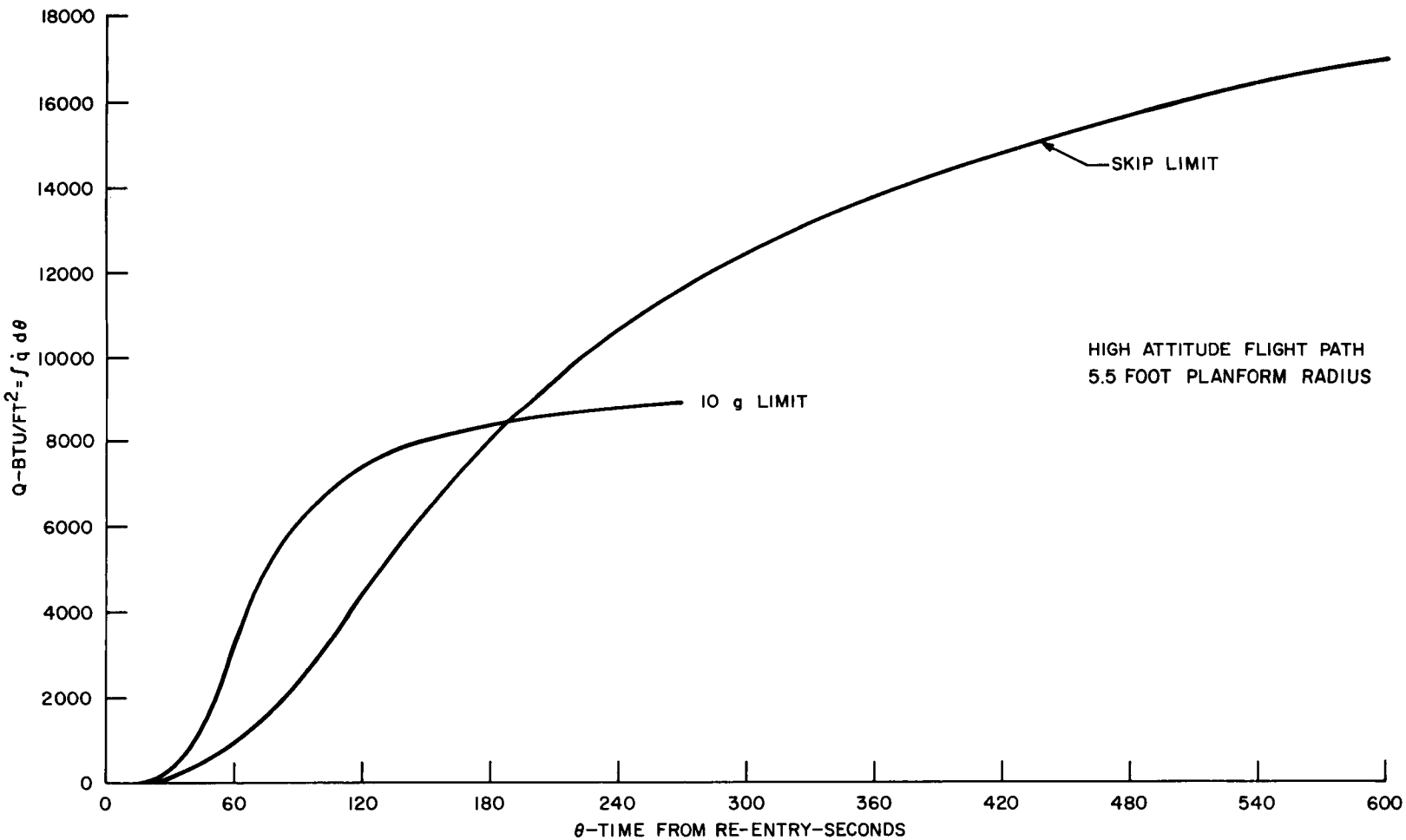


Figure XVI-3-20. Total heat per unit area versus time

corresponding to maximum stagnation enthalpy encountered during the flight, insofar as the variation of heat of ablation with velocity is presently known. The variation of velocity throughout the flight is accounted for, in computing required weight of ablative material*. No allowance is made for excess, or reserve, ablative material in any of the weights presented, and it can therefore be anticipated that practical considerations will cause an increase in the weights of systems involving ablation.

Many combinations of heat protection methods and materials were considered in the present study; those shown here are representative samples.

A. ALL ABLATION

The all-ablation systems, Figures XVI-3-21 (cooled) and XVI-3-22 (uncooled) include sufficient Teflon to absorb all the heat applied during the skip limit flight path, the flight path which causes maximum heating. The foamed silicon carbide is included to restrict the temperature of the aluminum primary structure to 200 F. Before an all-ablation system can be seriously considered, it will be necessary to accumulate experimental evidence which assures ablation during the entire heating period without structural deterioration of the ablative material.

B. ABLATION WITH RADIATION

Both cooled and uncooled ablative-radiative thermal protection combinations are shown In Figures XVI-3-23 through XVI-2-28. Either a structural insulation or a combination of a metallic outer wall panel and a high efficiency insulation protect the aluminum substructure from the long time, moderate heating; the teflon protects the aforementioned intermediate layer from the brief, but extreme, heating. The amount of ablative material required depends on the temperature capability of the intermediate layer, which, for zirconium oxide, coated molybdenum, and Haynes Alloy 25 is taken to be 4000 F, 2700 F, and 2000 F, respectively, i.e. these systems become radiative at a point on the heat flux-time curve corresponding to the temperature capability of the outer surface of the material under the teflon.

*BAC Report No. 7-60-941003

COOLED

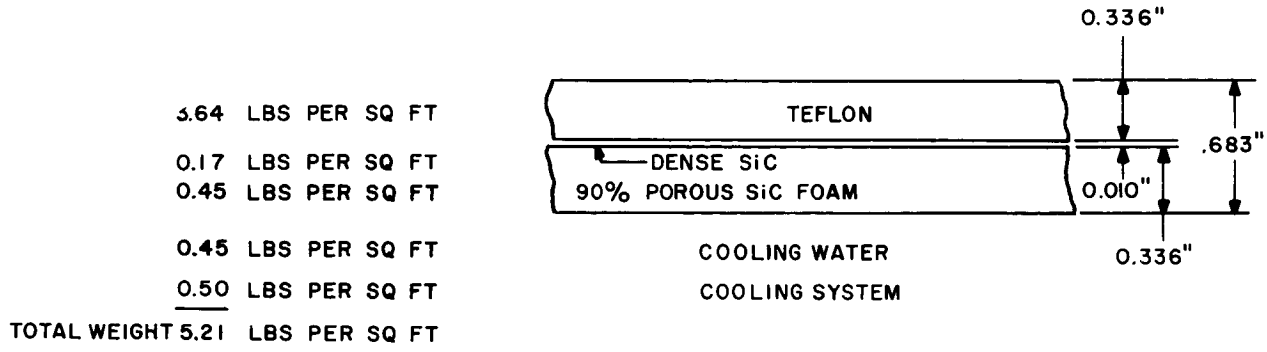


Figure XVI-3-21. Teflon on SiC foam (all heat absorbed by ablation)

UNCOOLED

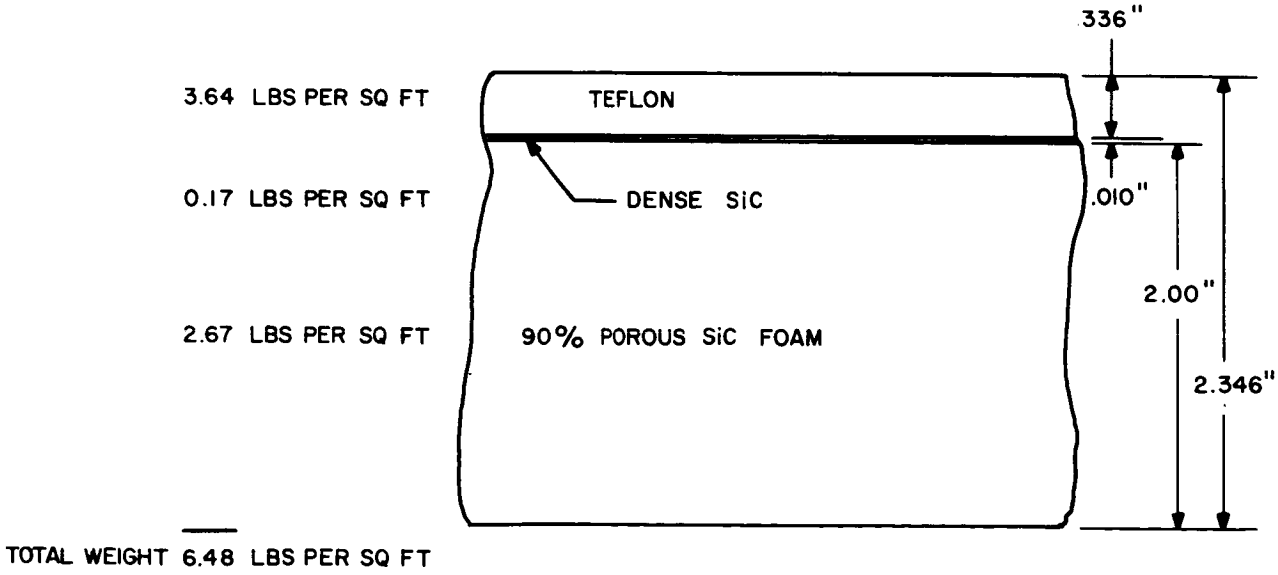


Figure XVI-3-22. Teflon on SiC foam (all heat absorbed by ablation)

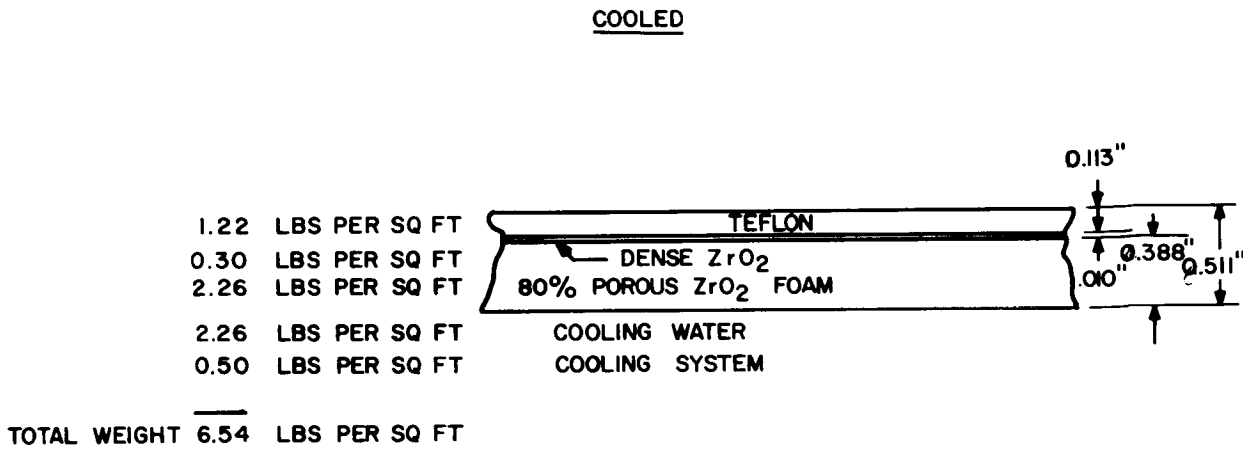


Figure XVI-3-23. Teflon on ZrO_2 foam (Teflon to $\dot{q}_{ZrO_2}$ at 4000 F)

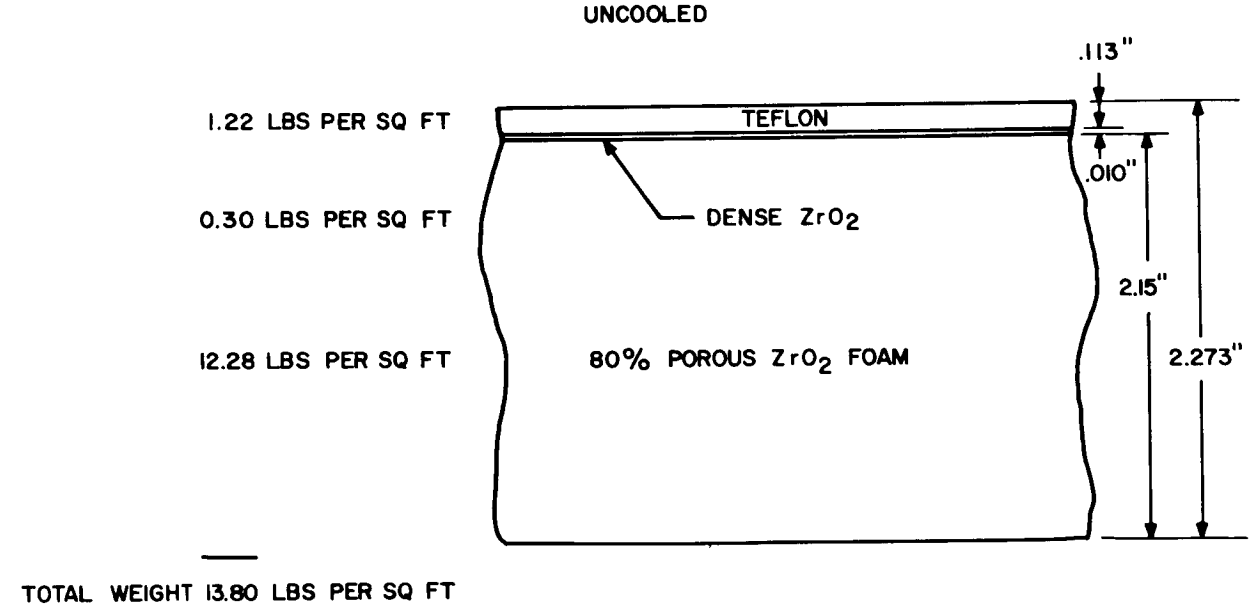


Figure XVI-3-24. Teflon on ZrO_2 foam (Teflon to $\dot{q}_{ZrO_2}$ at 4000 F)

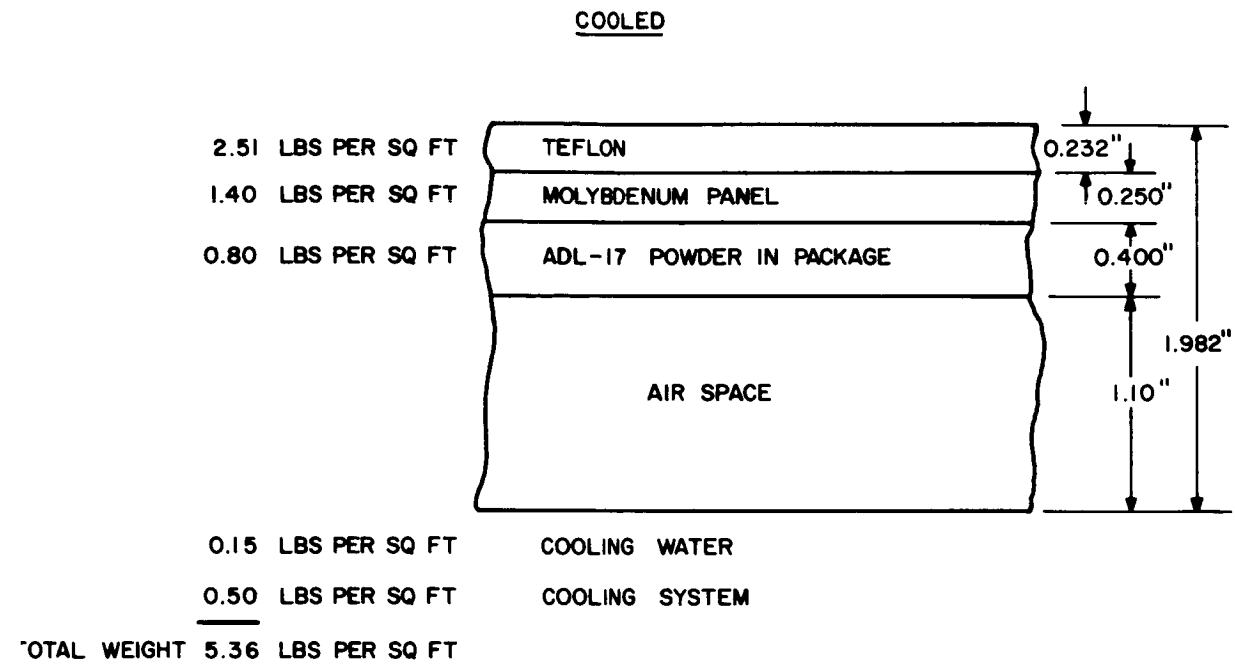


Figure XVI-3-25. Teflon on molybdenum double wall (Teflon to $\dot{q}$ molybdenum at 2700 F)

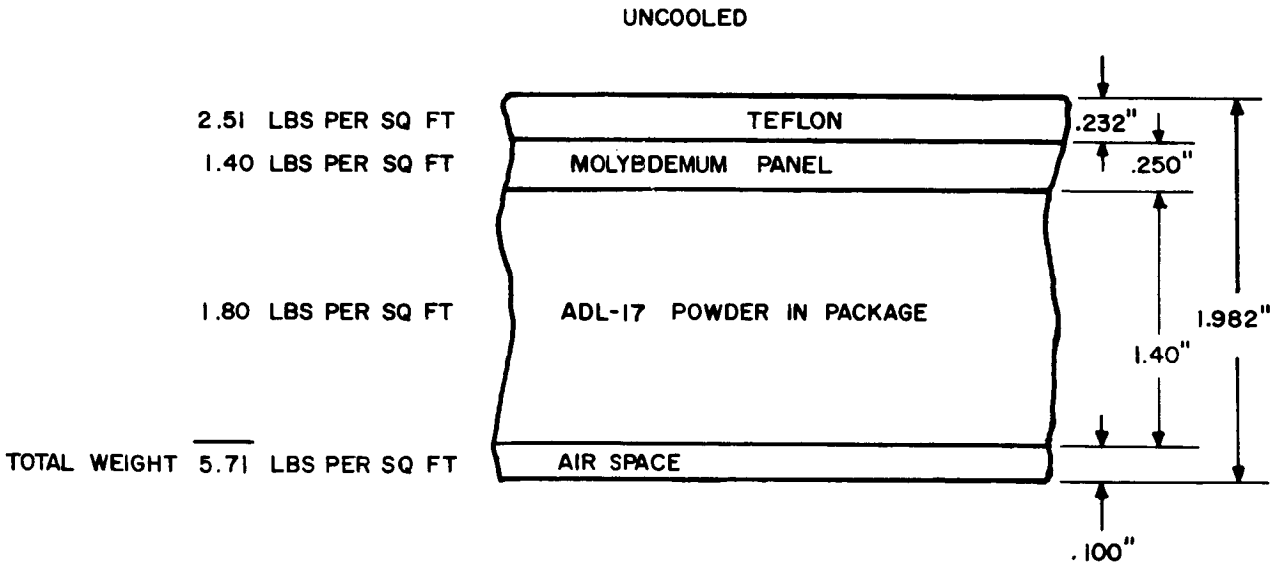


Figure XVI-3-26. Teflon on molybdenum double wall (Teflon to $\dot{q}$ molybdenum at 2700 F)

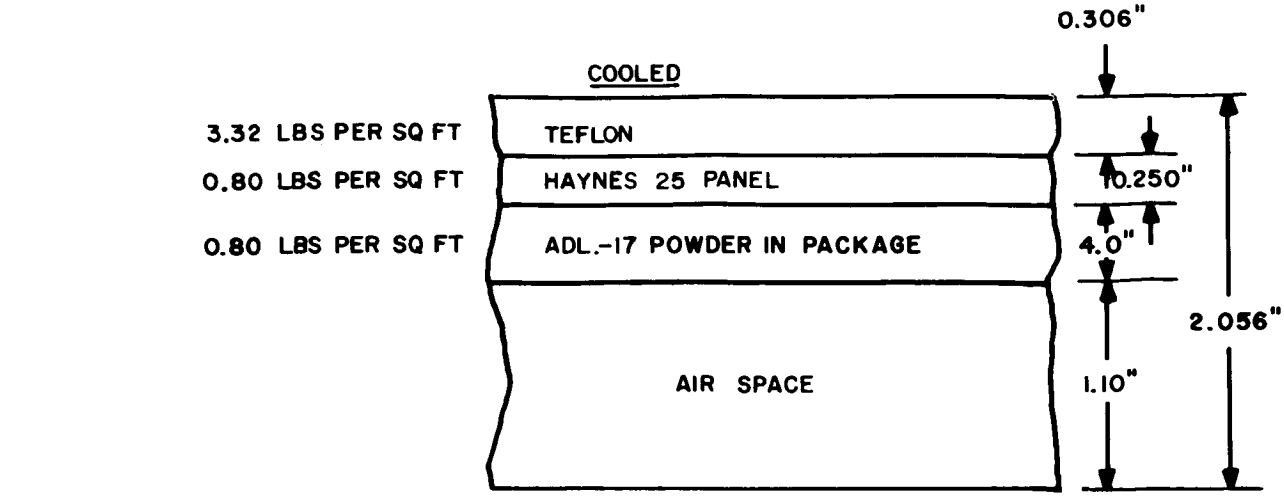


Figure XVI-3-27. Teflon on Haynes 25 double wall (Teflon to $\dot{q}$ Haynes at 2000 F)

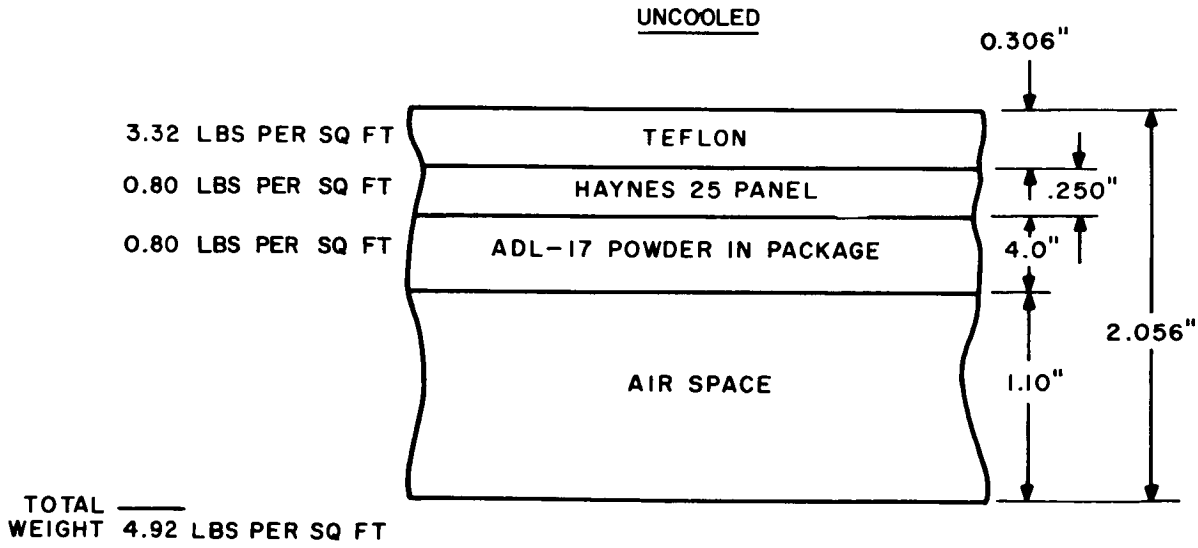


Figure XVI-3-28. Teflon on Haynes 25 double wall (Teflon to $\dot{q}$ Haynes at 2000 F)

C. ALL RADIATION

Figures XVI-3-29 through XVI-2-33 show cooled and uncooled systems which rely primarily on radiation to dissipate heat. Quotation marks are used because the loss of a very small amount of surface material is anticipated, although not enough to consider the process one of ablation. Experimental work with silicon carbide at Langley Research Center*, Chicago Midway Laboratories**, and Bell Aerosystems Company*** has shown that while heat fluxes greater than 100 Btu/sq ft-sec will oxidize silicon carbide, the rate at which material is lost is so low, that material loss for the length of heating period and heat flux level considered here would be insignificant. The silicon carbide acts as a radiator and insulator to protect the metal outer wall panel from temperatures exceeding its capability, the limit for tungsten being taken as 3200 F, with the molybdenum and Haynes 25 limits at 2700 F and 2000 F, as before. The all-"radiative" systems appear to provide the lowest weight for thermal protection.

A thermal protection system summary chart is included which shows the weight per square foot and over-all thickness for each of the thirteen thermal protection systems discussed.

Finally, a combined primary structure and thermal protection system summary chart Figure XVI-3-34 is presented which shows a progressive decrease in total airframe weight as a result of technological improvements.

Figure XVI-3-35 summarizes the weights of the various thermal protection systems investigated.

Figure XVI-3-36 is the combined weight summary of both the structure and the thermal protection.

---

* NASA TN D-644
** WADD TR 59-87
*** BAC D143-941-024

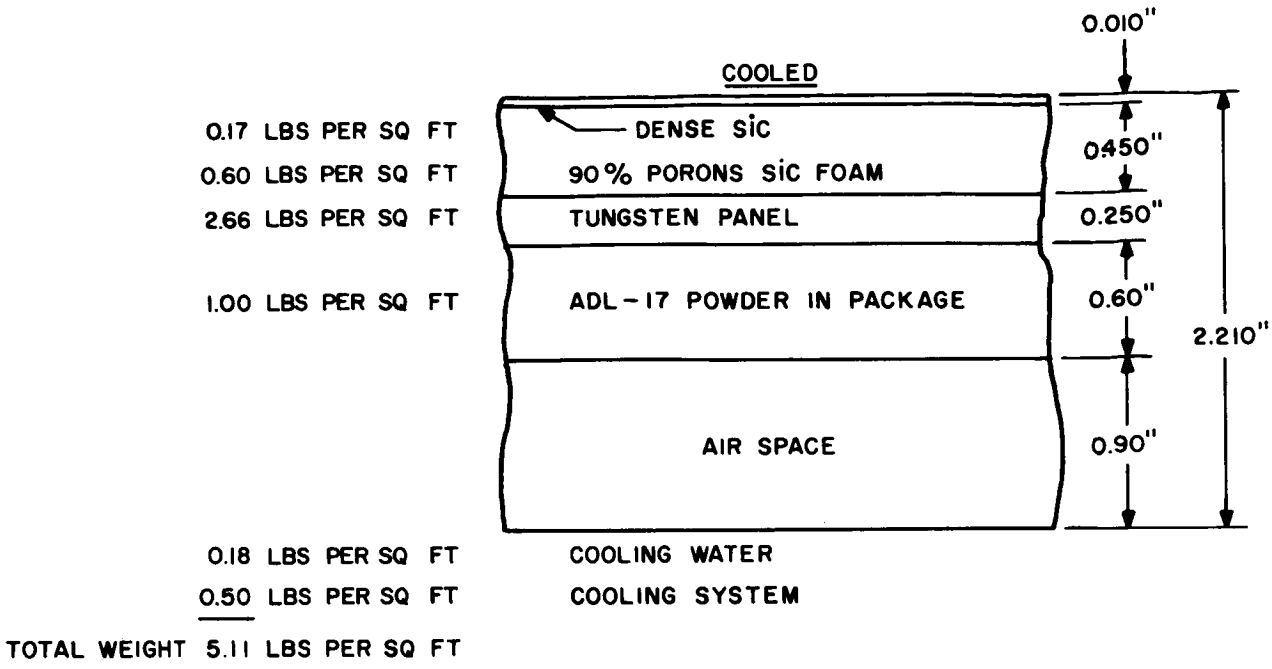


Figure XVI-3-29. SiC on tungsten double wall (SiC protects tungsten to 3200 F)

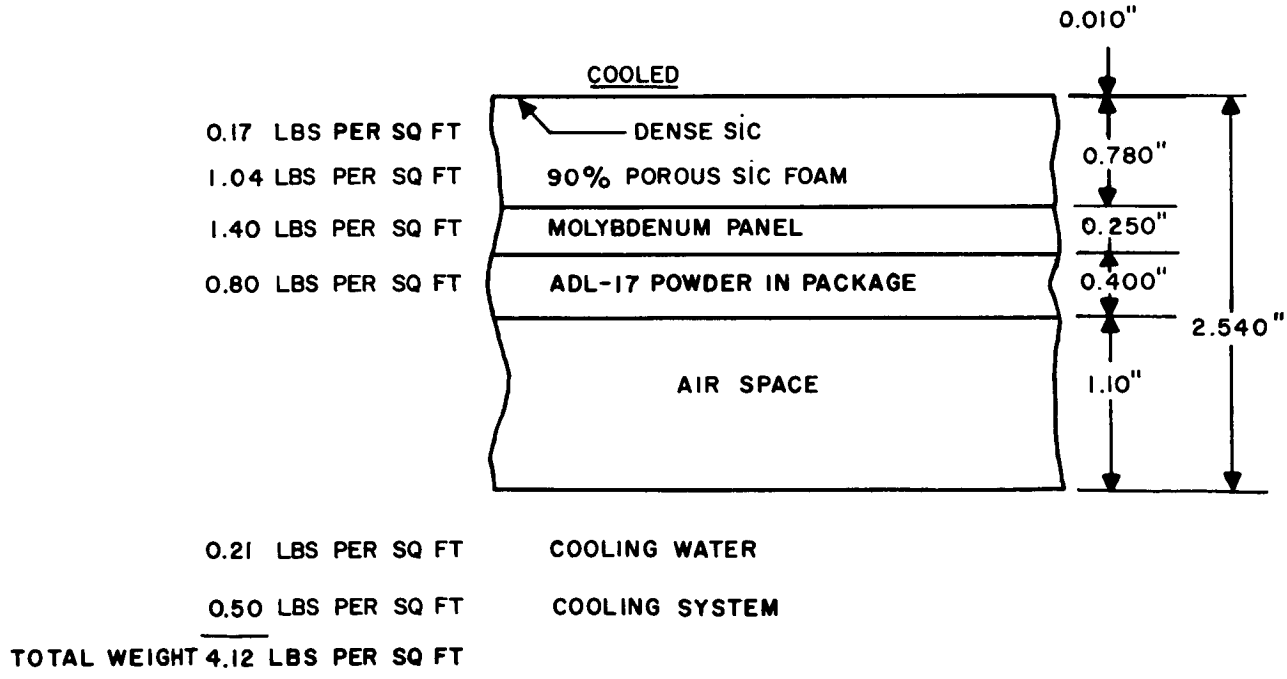


Figure XVI-3-30. SiC on molybdenum double wall (SiC protects molybdenum to 2700 F)

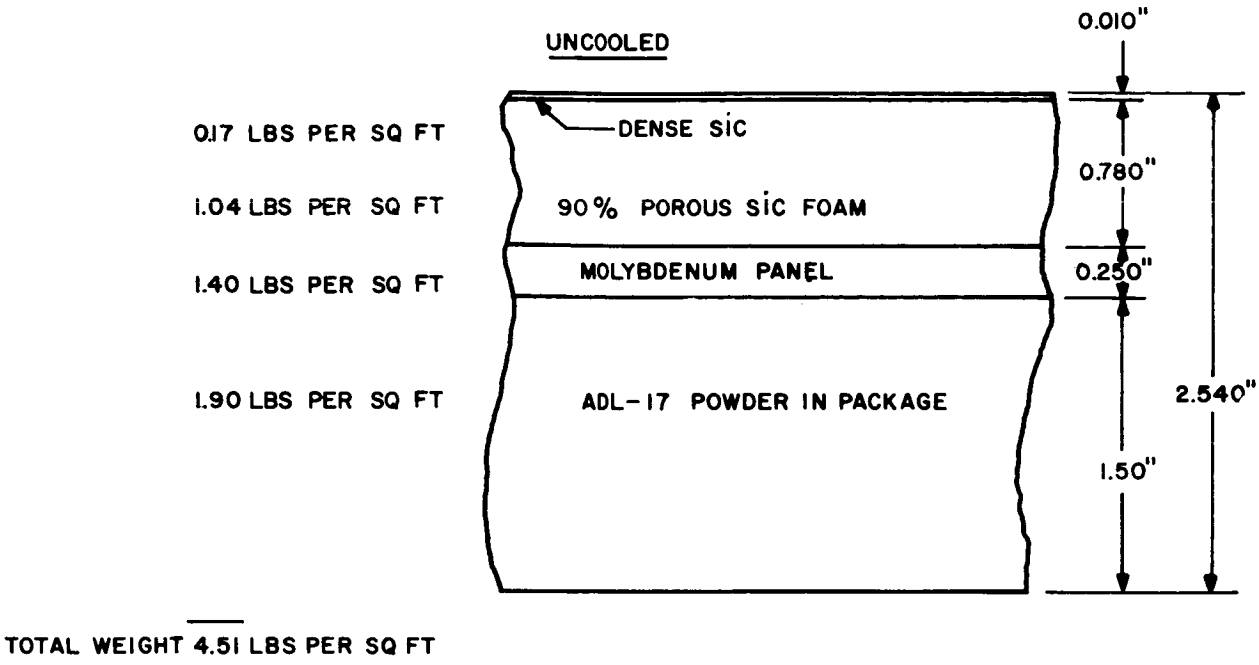


Figure XVI-3-31. SiC on molybdenum double wall (SiC protects molybdenum to 2700 F)

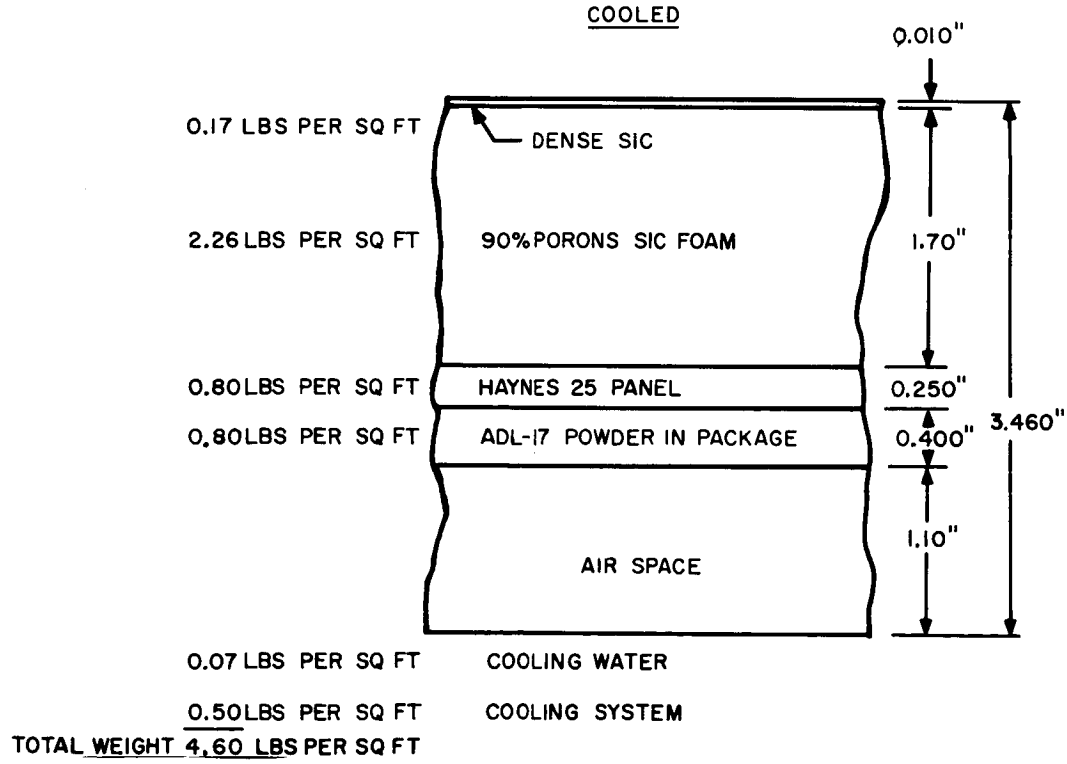


Figure XVI-3-32. SiC on Haynes 25 double wall (Sic protects Haynes 25 to 2000 F)

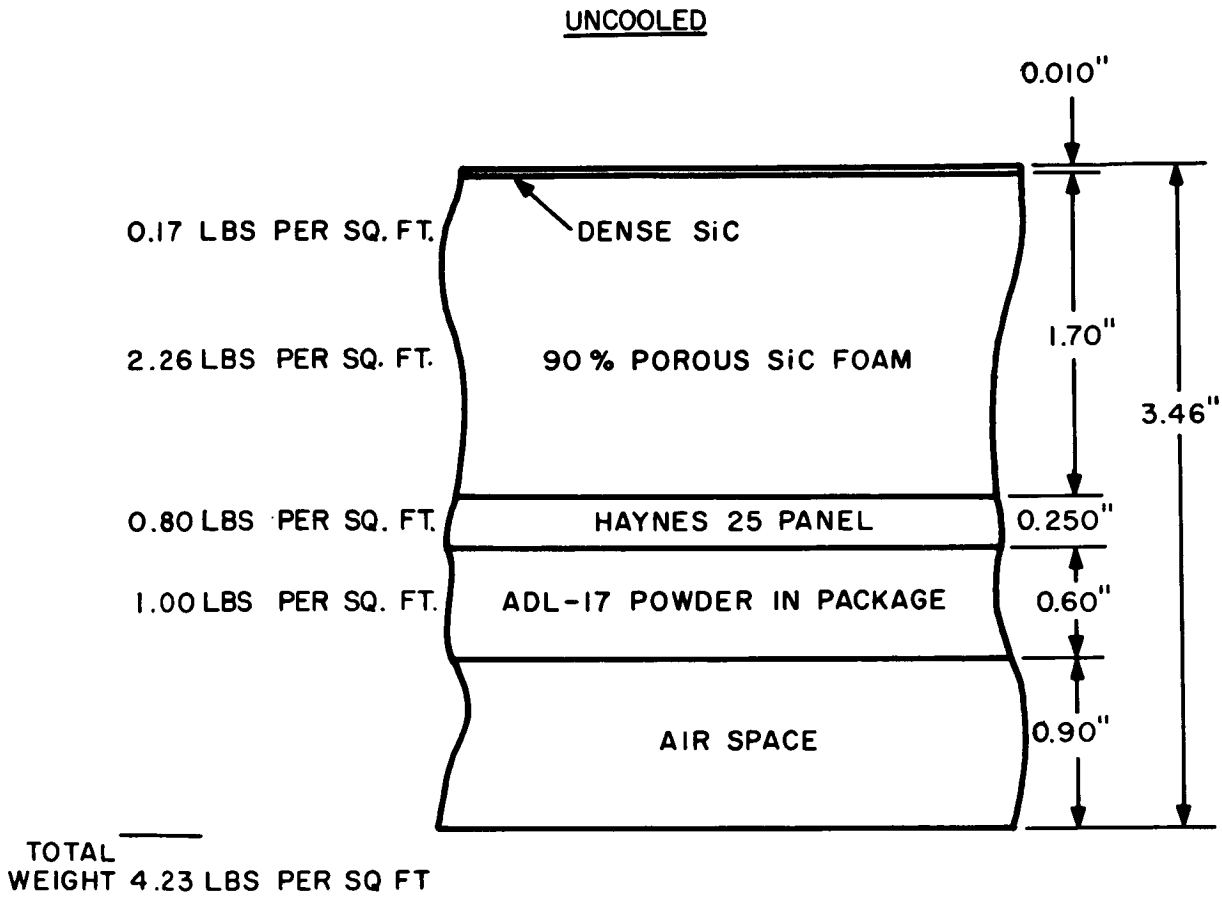


Figure XVI-3-33. SiC on Haynes 25 double wall (SiC protects Haynes 25 to 2000 F)

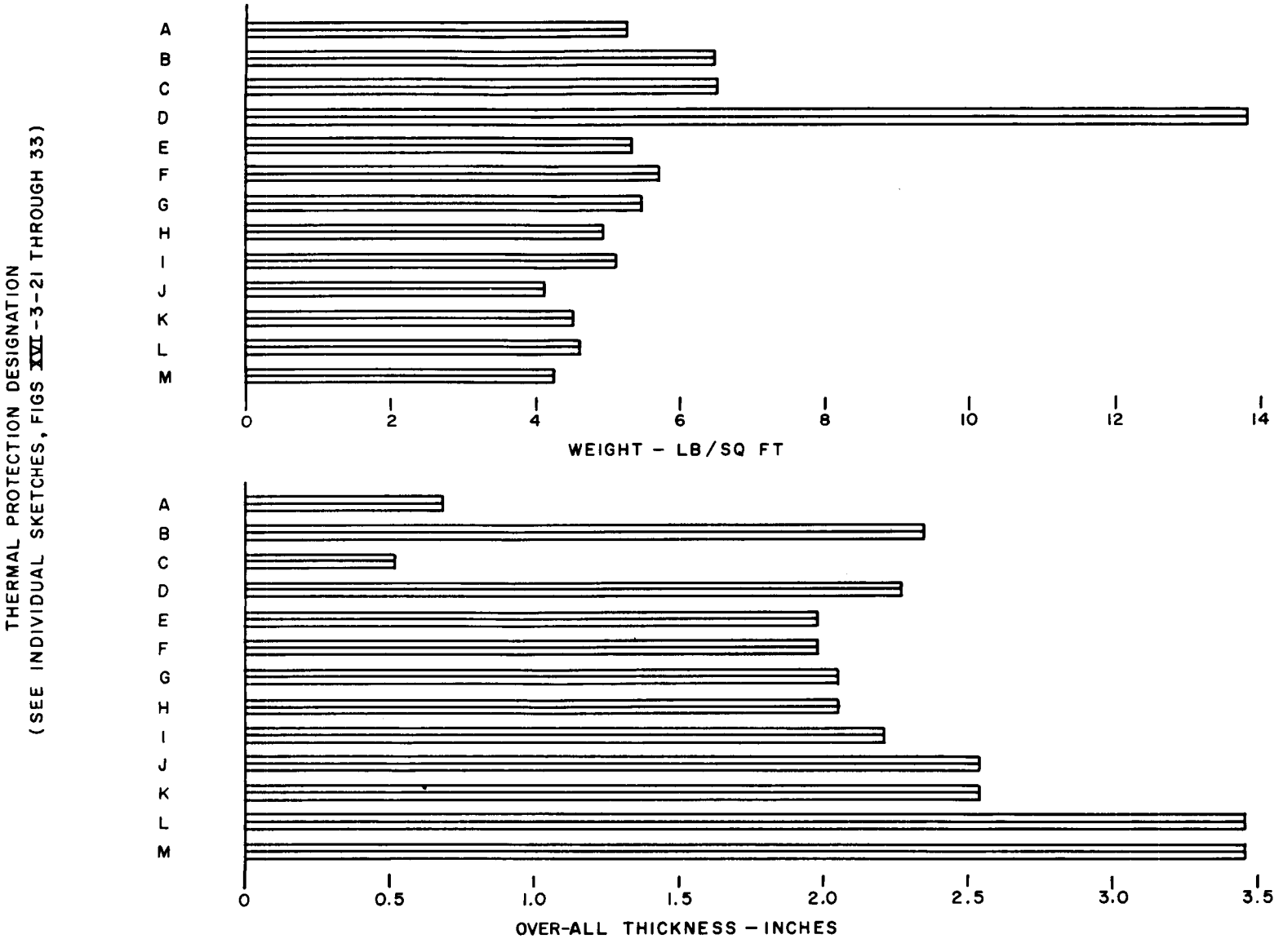


Figure XVI-3-34. Thermal protection and thickness summary (excluding aluminum primary structure)

CONFIDENTIAL

CONFIDENTIAL

ALL ABLATION

Ablative Material
Structural Insulation
Aluminum Substructure

Teflon on Silicon Carbide Foam - 5.21 psf.
GE 500 Series --------

ABLATION WITH RADIATION

Ablative Material
High-Temperature Structural Insulation or High-Temperature Structural Material and High-Efficiency Insulation
Aluminum Substructure

COOLED

Teflon on Zirconium Oxide Foam - 6.54 psf.
Teflon on Molybdenum Double Wall - 5.36 psf.
Teflon on Haynes 25 Double Wall - 5.46 psf.

UNCOOLED

Teflon on Zirconium Oxide Foam - 13.80 psf.
Teflon on Molybdenum Double Wall - 5.71 psf.
Teflon on Haynes 25 Double Wall - 4.92 psf.

ALL "RADIATION"

High-Temperature Radiative Material
High-Temperature Structural Material
High-Efficiency Insulation
Aluminum Substructure

COOLED

Silicon Carbide Foam on Tungsten Double Wall - 5.11 psf.
Silicon Carbide Foam on Molybdenum Double Wall - 4.12 psf.
Silicon Carbide Foam on Haynes 25 Double Wall - 4.60 psf.

UNCOOLED

Silicon Carbide Foam on Tungsten Double Wall - _____ *
Silicon Carbide Foam on Molybdenum Double Wall - 4.51 psf.
Silicon Carbide Foam on Haynes 25 Double Wall - 4.23 psf.

* Not feasible with present Double Wall configuration and aluminum primary structure.

Figure XVI-3-35. Weight summary

Configuration / Item		Modified Lenticular R-3 Conventional Aluminum Construction		Improved R-3 Configuration (See Figure XVI-2-16 for Wetted Areas) Pressure-Supported Aluminum Ellipsoid of Revolution Structure		Improved R-3 Configuration Beryllium Material Used For All Preliminary Structure Except Outer Shell	
Primary Structure	Wetted Area Ft^2	Unit Weight #/Ft^2	Total Weight #	Unit Weight #/Ft^2	Total Weight #	Unit Weight #/Ft^2	Total Weight #
Elevon	30.0	2.00	60.0	2.00	60.0	1.34	40.0
Tail	40.0	2.55	102.0	2.55	102.0	1.62	65.0
Body-Nose	50.2	1.91	96.0	2.00	626.0	1.70	530.0
-Cockpit	174.7	3.60	630.0				
-Equip. Comp.	106.3	4.00	425.0	3.00	320.0	2.34	250.0
Total Primary Structure			1313.0		1108.0		885.0
Thermal Protection							
Elevon-Lower Surface	15.0	5.70 (Teflon on Molybdenum Double Wall)	85.5	4.25 (Silicon Carbide on Haynes 25 Double Wall)	63.5	3.75 (Silicon Carbide Sandwich)	56.0
Fin-Outer Surface	20.0	3.00 (Molybdenum Double Wall)	60.0	3.00 (Molybdenum Double Wall)	60.0	3.00 (Molybdenum Double Wall)	60.0
-Inner Surface	20.0	2.25 (Haynes 25 Double Wall)	45.0	2.25 (Haynes 25 Double Wall)	45.0	2.25 (Haynes 25 Double Wall)	45.0
Body-Lower Surface	174.0	5.70 (Teflon on Molybdenum Double Wall)	992.0	4.25 (Silicon Carbide on Haynes 25 Double Wall)	740.0	3.75 (Silicon Carbide Sandwich)	652.0
-Upper Surface	175.0	2.25 (Haynes 25 Double Wall)	394.0	2.25 (Haynes 25 Double Wall)	394.0	2.25 (Haynes 25 Double Wall	394.0
Total Thermal Protection			1576.5		1302.5		1207.0
TOTAL WEIGHT			2889.5		2410.5		2092.0

Figure XVI-3-36. Combined weight summary — primary structure and thermal protection

WEIGHT ANALYSIS

Presented in Table XVI-3-1 is a weight summary and in Table XVI-3-2 is a detailed weight breakdown of the R-3 modified lenticular configuration. The center of gravity is located at 82 inches from the leading edge.

In table XVI-3-3 is a slightly modified (updated) weight summary of the present design capability weight and the predicted 1963 capability weight. In Table XVI-3-2 is the updated weight breakdown between the R/V and the S/V.

TABLE XVI-3-1. WEIGHTS SUMMARY OF R-3 CONFIGURATION

Item	Present	Weight lbs.
Recovery Vehicle		6447.0
Structure	3442.0	
Inst. & Comm	101.0	
Power Supply	210.0	
Life Support	1135.0	
Search & Recovery	481.0	
Nav. & Control	725.0	
Elec.	353.0	
Space (Mission) Module		13360.0
Structure	2252.0	
Instr. & Control	60.0	
Power Supply	700.0	
Life Support	635.0	
Nav. & Control	500.0	
Elec.	320.0	
Prop. Sys.	8893.0	
Subtotal		19807.0
Less 92% of Abort Rockets		-828
Total Gross Weight		18,979

Detailed Weight Breakdown

TABLE XVI-3-2. DETAIL WEIGHT OF R-3 CONFIGURATION

Structure		
Recovery Vehicle		3442
Shield (Ablative & Outer Wall Panels)	1386.0	
Skin	212.9	
Pressure Bulkheads	140.0	
Longerons	135.8	
Frames & Bulkheads	397.9	
Fin Pylon	74.0	
Landing Gear Attach.	30.0	
Elevon Attach.	16.0	
Escape Attach.	16.0	
Booster Attach.	35.0	
Misc. Angles, Fittings, Etc.	39.4	
Tunnel Including Press. Door	54.0	
Elevons (Complete)	146.0	
Vertical Tail (Complete)	472.0	
Supports	96.0	
Windshield	71.0	
Windshield Cowl & Jettison System	77.0	
Hatch Install	43.0	
Space (Mission) Vehicle		2252
Mission Module	726.0	
Fairings Fwd. & Aft Module	960.0	
Tunnel	31.0	
Attaching Structure	535.0	
Total Structure		5694
Instrument and Communications		
Recovery Vehicle		101.0
2KMC - 2 Watt Dr. Xmtr.	15.0	
400 MC - 3 Watt Xmtr. Rec.	5.0	
Data Recorder	25.0	
Multiplexer	4.0	
PMC Multiplexer	38.0	
400 MC 3 Watt Xmtr	6.0	
Antenna (2 required)	8.0	

TABLE XVI-3-2. DETAIL WEIGHT OF R-3 CONFIGURATION (Continued)

Space (Mission) Vehicle		60.0
2KMC 16 Watt Power Amp.	15.0	
2KMC Rec. Decoder	10.0	
T V Camera	17.0	
Photographic Camera	2.0	
Antenna	16.0	
Power Supply System		
Recovery Vehicle		210.0
Batteries	210.0	
Space (Mission) Vehicle		700.0
Batteries	350.0	
Solar Cell Reflector	350.0	
Search and Recovery		
Recovery Vehicle		481.0
Beacon — Elec.	8.0	
Beacon — Antenna	3.0	
Chaff	8.0	
Flare Cluster	9.0	
Power Supply	15.0	
Programmer	6.0	
Chaff	2.0	
Light	2.0	
So-Far Bomb	4.0	
Dye Marker	4.0	
Main Chute	130.0	
Chute Equipment	40.0	
Impact System		240.0
Hand Oper. Devices	10.0	
Propulsion System		
Space (Mission) Vehicle		8893.0
Abort System	900.0	
Thrust Chamber System (Tanks Included)	2473.0	
Fuel, Oxidizer and Press. Gas	5520.0	
Navigation and Control		
Recovery Vehicle		725.0
Recovery Flight Control System	223.0	
Stable Platform (Including Astrotracker and 3 Accelerometers)	75.0	

TABLE XVI-3-2. DETAIL WEIGHT OF R-3 CONFIGURATION (Continued)

Platform Elec.	15.0	
Computer	50.0	
Control Console (Instr.)	62.0	
Radar Altimeter	50.0	
Navigation Aids	25.0	
IR Sensor and Elec. (2 required)	26.0	
Miscellaneous	24.0	
Hydraulic System	175.0	
Space (Mission) Vehicle		500.0
Propellant	500.0	
Electrical System		
Recovery Vehicle		353.0
Elec. Wiring	76.0	
Elec. Misc. Equipment	37.0	
Miscellaneous	20.0	
Motors	130.0	
Inverters	90.0	
Mission (Space) Vehicle		320.0
Elec. Wiring	100.0	
Elec. Misc. Equipment	136.0	
Umbilical System	20.0	
Separation	20.0	
Miscellaneous	44.0	
Life Support Equipment		
Recovery Vehicle		1135.0
Crew (3)	540.0	
Acc. Seats & Restraint System	90.0	
Galley & Food (Emergency) (Incl. H_2O)	75.0	
Survival Equipment	75.0	
Personal Equipment	30.0	
Console (Instr.)	50.0	
Lighting & Trim	5.0	
Miscellaneous Furnishings	25.0	
Cabin Comm.	5.0	
O_2 Emergency	50.0	
Diluent Supply	28.0	
CO_2 Rem. (Normal & Emergency)	70.0	
Heat Exchanger (Emergency)	10.0	
Fire Control	15.0	
Sen & Control	25.0	

TABLE XVI-3-2. DETAIL WEIGHT OF R-3 CONFIGURATION (Continued)

Urine & Feces Storage	10.0	
Water Recovery	10.0	
Ion Control	5.0	
Miscellaneous	27.0	
Space (Mission) Vehicle		635.0
Space Radiators	35.0	
Crew Stat. Seat	5.0	
Toilet	25.0	
Galley & Good (Water Included)	247.0	
Water & Container	25.0	
Instr. (Console)	20.0	
Recreation Equipment	1.0	
Lighting and Trim	5.0	
Miscellaneous Furnishings	15.0	
O_2 Primary	150.0	
Diluent Supply	28.0	
Heat Exchanger (Prim.)	20.0	
Fire Control	15.0	
Maintenance Equipment (25 & Misc. 19)	44.0	

Weight Summary, R-3 Configuration

TABLE XVI-2-3. WEIGHT SUMMARY, R-3 CONFIGURATION

Sub System	Present Design Capability Weight	Predicted 63 Capability Weight
Propulsion	8600	8100
Life Support	1909	1700
Nav & Control	1150	900
Search & Recovery	481	447
Power Supply	782	630
Electrical System	673	478
Instr. & Comm.	159	150
Structure	4789	3589
Gross Weight	18543	15994

Weight Breakdown, R3 Configuration

TABLE XVI-3-4. WEIGHT BREAKDOWN, R-3 CONFIGURATION

Sub System	R/V	S/V
Structure	3100	1689
Instr. & Comm.	99	60
Power Supply	357	425
Search & Recovery	481	-
Propulsion Sys.	-	8600
Nav. & Control	725	425
Elect. Sys.	353	320
Life Support	1358	551
Total	6473	12,070

Gross Wt. R/V + S/V = 18,543
Abort Wt. = R/V Wt. = 6473

SUMMARY

To summarize, the following points favor this configuration.

The modified lenticular re-entry vehicle having a hypersonic L/D of 0.7, provides a wide re-entry corridor, thus compensating for guidance errors and propulsion deficiencies. The high L/D likewise, provides large cross-range and down-range maneuverability. The conventional landing assures a recovery at a preselected site and permits possible reuse of the vehicle. The on-the-pad abort system provides a minimum of separation sequencing, a minimum of abort weight, and the capability of recovery of the vehicle on the nearby skid strip or beach. All accelerations are applied to the crew in the most favorable direction (eyeballs in).

These points justify continued investigation and preliminary design of the R-3 configuration for the APOLLO mission.

4. COMPARISON OF D-2 AND R-3 VEHICLES

Characteristics of the two selected vehicles are summarized in Tables XVI-4-1 and XVI-4-2, with the following features most significant:

1. The modular D-2 vehicle represents a present design capability, and the lightest weight arrangement of all vehicles studied.
2. The D-2 vehicle arrangement is more compatible with the booster due to the use of a uniform nonocoque S/V structure having no abrupt cross-section area changes.
3. Both vehicles have potential for orbital operations such as rendezvous.
4. The R-3 vehicle has better landing and recovery features, and has a local obstacle avoidance capability.
5. The R-3 has a wider re-entry corridor which compensates for guidance and propulsion inaccuracies (i.e. re-entry angle).
6. The R-3 has a simpler abort sequence in that only the R/V is separated from the booster.

TABLE XVI-4-1. CONFIGURATION COMPARISON

DESIGNATION S/V TYPE R/V TYPE	B2 MODULAR SEMI-BALLISTIC MK 2	C2 MODULAR SEMI-BALLISTIC RVX 2	D2 MODULAR SEMI-BALLISTIC MODIFIED NERV/ DISCOVERER	R1 INTEGRAL HIGH L/D DYNASOAR	R3 MODULAR MODIFIED LENTICULAR
L/D-HYPERSONIC	0.29	0.7	0.6	2.0	0.7
SUBSONIC-CLEAN	—	—	—	4.4	7.0
SUBSONIC-GEAR DOWN	—	—	—	3.8	5.5
HYPERSONIC W/CLA AT MAXIMUM L/D	243	265	95	200	66
PREDICTED 63 WEIGHT-R/V	4780	7060	4258	9605	5559
-TOTAL	16,500	20,000	14,980	20,000	15,994
ALLOWABLE RE-ENTRY CORRIDOR	±1.3°	±2.0°	±1.8°	±2.4°	±2.0°
NOMINAL PULLOUT G'S	12	12	12	12	12
CROSS RANGE MANEUVER-NM	100	1000	700	3000	1000
TYPE LANDING-LAND	CHUTE	CHUTE	CHUTE	CONVENTIONAL	CONVENTIONAL
-WATER	CHUTE	CHUTE	CHUTE	CONVENTIONAL OR CHUTE	CONVENTIONAL OR CHUTE
-OFF PAD ABORT	CHUTE	CHUTE	CHUTE	CONVENTIONAL OR CHUTE	CONVENTIONAL OR CHUTE
MODULES ABORTED	R/V+M/M	R/V+M/M	R/V+M/M	R/V ONLY	R/V ONLY
METEOROID/RADIATION PROTECTION	5.4#/FT^2	12.9#/FT^2	6.0#/FT^2	6.0#/FT^2	5.46#/FT^2
CREW SPACE	FAIR	POOR	GOOD	GOOD	GOOD
FAIL SAFE RE-ENTRY	FAIR	POOR	FAIR	POOR	POOR
DESIGN FLEXIBILITY	GOOD	GOOD	GOOD	FAIR	FAIR
LOCAL OBSTACLE AVOIDANCE	POOR	POOR	POOR	GOOD	GOOD
EARLY DEVEL. CAPABILITY	GOOD	GOOD	GOOD	FAIR	FAIR

CONFIDENTIAL
CONFIDENTIAL

TABLE XVI-4-2. COMPARISON OF RECOMMENDED CONFIGURATIONS

	D-2	R-3
Cross Range Maneuver (High L/D)	+	+
Weight	+	-
Horizontal Landing	-	+
Low Impact Loads	-	+
Obstacle Avoidance	-	+
Sense of Pilot Control	-	+
Crew Space	+	+
Meteoroid/Radiation Protection	+	-
Fail Safe Re-Entry	+	-
Re-Entry Corridor	+	+
Early Development Capability	+	-